REPORT NUMBER 150

SEPTEMBER 1964

CALCULATED INSTA
OWERPLANT PERFOR

XV-5A

LIFT FAN FLIGHT RESEARCH AIRCRAFT

CONTRACT NUMBER DA44-177-TC-

GENERAL ELECTR

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REPORT NUMBER 150

Calculated Installed
Power Plant Performance

XV-5A Lift Fan
Flight Research Aircraft Program

September 1964

Advanced Engine and Technology Department
General Electric Company
Cincinnati, Ohio 45215

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1.0 SUMMARY

This report presents calculated installed performance characteristics for the U. S. Army XV-5A lift fan research aircraft propulsion system. The propulsion system consists of two General Electric X353-5B powerplants, one G.E. X376 pitch control fan, and associated ducting, controls and accessory equipment.

Installed performance of turbojet mode is presented for ARDC Standard Day and ANA 421 Hot Day for one and two engine operation. Performance data include gross thrust, propulsion system drag, net thrust, fuel flow and cooling system drag. A sea level static thrust of 4,920 pounds is estimated for an ARDC Standard Day. For ANA 421, Hot Day conditions at 2,500 feet altitude, static thrust is 4,250 pounds.

Performance of the XV-5A propulsion system in the lift fan mode is treated as an integral part of the performance of the total airplane system. The estimate for airplane lift fan mode performance was based on:

1. Static (zero speed) installed performance data supplied by General Electric. This data was used to represent the aircraft for out-of-ground effect.

2. Special aerodynamic propulsion system coefficients derived from full and small scale wind tunnel model test results. These aero-propulsion data, applicable to in-ground effect and transition flight operations, are presented in other XV-5A Reports. For the hovering condition, the total installed untrimmed lift without allowance for pitch or roll control requirements is 15,020 pounds for standard sea level day, and 12,625 pounds for ANA hot day at 2,500 foot altitude.

A detailed analysis of J85 engine operation at near idle condition (47% to 60% rpm) showed that exhaust gas temperature increased rapidly with increasing engine air inlet temperature and shaft power extraction. Thus, to preclude exceeding exhaust gas temperature limits, due to re-ingestion of hot engine exhaust gases and/or varying power extraction for system checkout, a minimum rpm of 70% for the J85 engines is recommended for XV-5A fan mode operation.

The engine air inlet shows excellent performance throughout its required operating envelope. A minimum total pressure recovery of 98.4% is available for static operation and the pressure recovery exceeds 99% for high speed cruise flight at Mach 0.7.

Recommendations also are made for research studies to investigate:

1. Methods of isolating and defining lift fan contributions to aircraft performance, as a function of the operating environment governed by aircraft geometrical and flight variables.

2. Factors affecting fan-induced flow fields near the ground which produce re-circulation of hot gas, particle impingement and re-ingestion, induced steady and unsteady forces, moments etc.

The propulsion system and its associated components, such as thrust spoilers, ejectors etc. are estimated to perform their intended functions adequately throughout the XV-5A flight envelope.

2.0 INTRODUCTION

Presented herein are the calculated installed powerplant performance characteristics of the U.S. Army XV-5A Lift Fan Research Aircraft. The XV-5A, designed and built under contract for the General Electric Company, by the Ryan Aeronautical Company, is a two-place V/STOL aircraft capable of flight at high subsonic Mach numbers, and designed for research flight testing of the General Electric lift fan propulsion system.

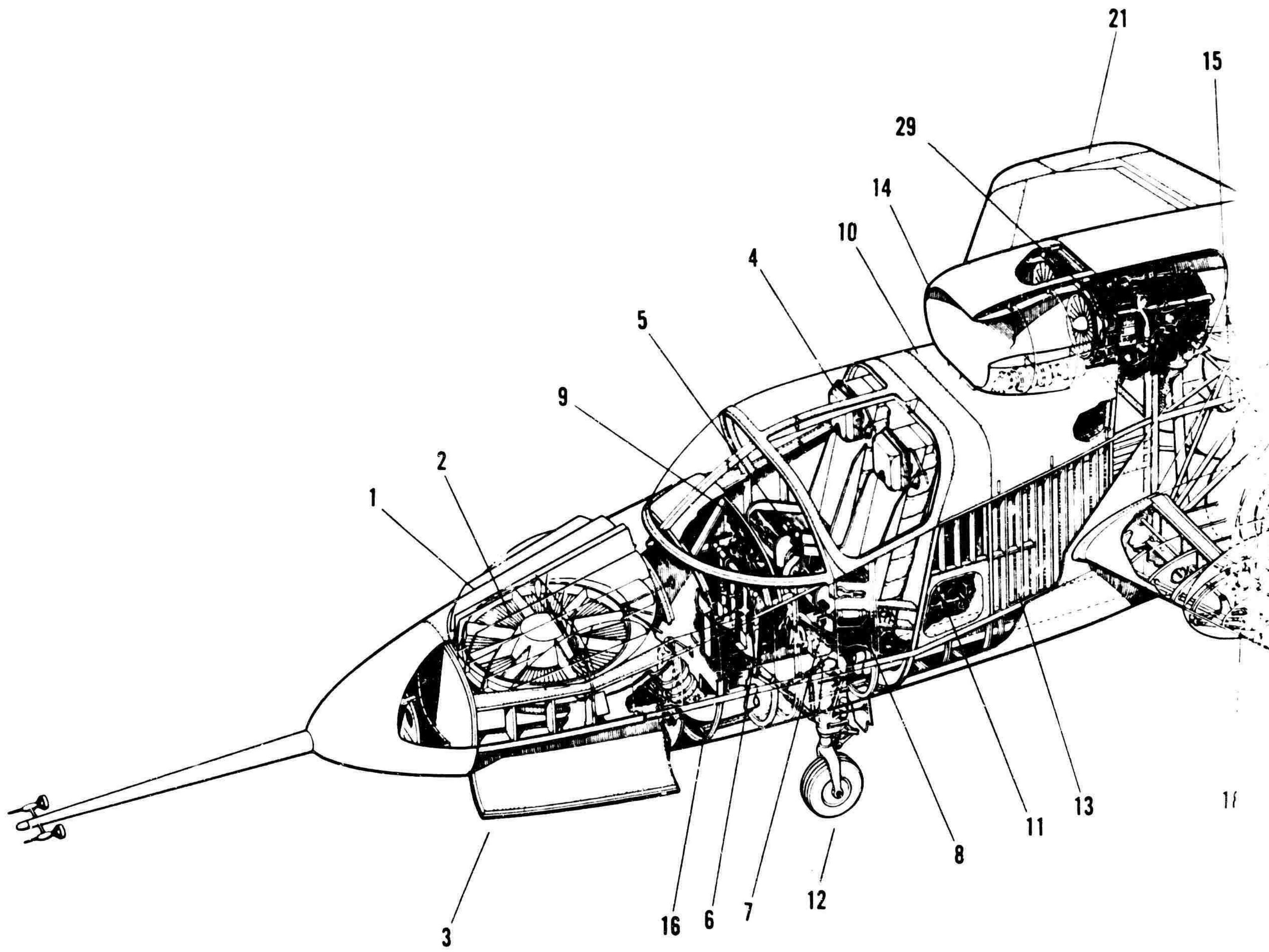
The XV-5A propulsion system consists of the following elements: a single engine air inlet with provisions for boundary layer bleed, two G.E. X353-5B lift fan propulsion systems including two J85-GE-5 gas generators modified for non-reheat operation, two gas diverter valves for selecting fan mode or turbojet mode operation, and two wing, tip-turbine lift fans, two wing fan inlet closure systems, one G.E. X376 tip-turbine pitch control fan, one pitch fan inlet closure system, and two pitch fan thrust reverser doors, two sets of wing-fan crossover and pitch fan ducting, and two exhaust duct systems, each including one tailpipe, cooling air shroud, nozzle, ejector and thrust spoiler. The relationship of these components is illustrated in Figures 2.1 and 2.2.

Each gas generator is directly connected to a gear box assembly at the power take-off pad; which drives a set of two cooling air blowers, one electrical generator, and one hydraulic oil pump. Each gas generator drives an air turbine driven fuel boost pump by means of compressor bleed air. Cross ducting and check valves are used to permit single engine operation of both boost pumps.

The gas power distribution system in fan mode operation is designed to provide balanced and proportionate power to the wing and pitch fans from each gas generator. The forward inboard and the aft inboard quadrants of the wing fan tip turbines are supplied gas power by the left and right gas generators respectively. Likewise, the left and right gas generators supply gas power to the aft-left and aft-right quadrants of the pitch fan tip turbine respectively. Thus, in the event of an engine-out condition,

aircraft balance is retained even though at reduced fan thrust levels. The wing fans counter-rotate. Viewed from above, the left wing fan rotates counterclockwise, and the pitch fan rotates clockwise. Viewed from the aft position, the gas generator turbines rotate counterclockwise.

In the turbojet mode, the gas generators act as conventional turbojet engines with fixed nozzles. Thrust spoiler blades may be extended into the exhaust streams, deflecting the streams symmetrically outward to reduce aircraft axial thrust, while maintaining high gas generator speeds in preparation for conversion from turbojet mode to fan mode flight. While in the turbojet mode, powerplant operation is restricted to the speed-altitude envelope shown in Figure 2.3, which was based on the X353-5B engine specification described in Reference 1 of this report. Engine performance data is included for 40,000 feet altitude (above the 36,089 foot specification limit) as it was derived using the digital computer card deck supplied by General Electric Company.



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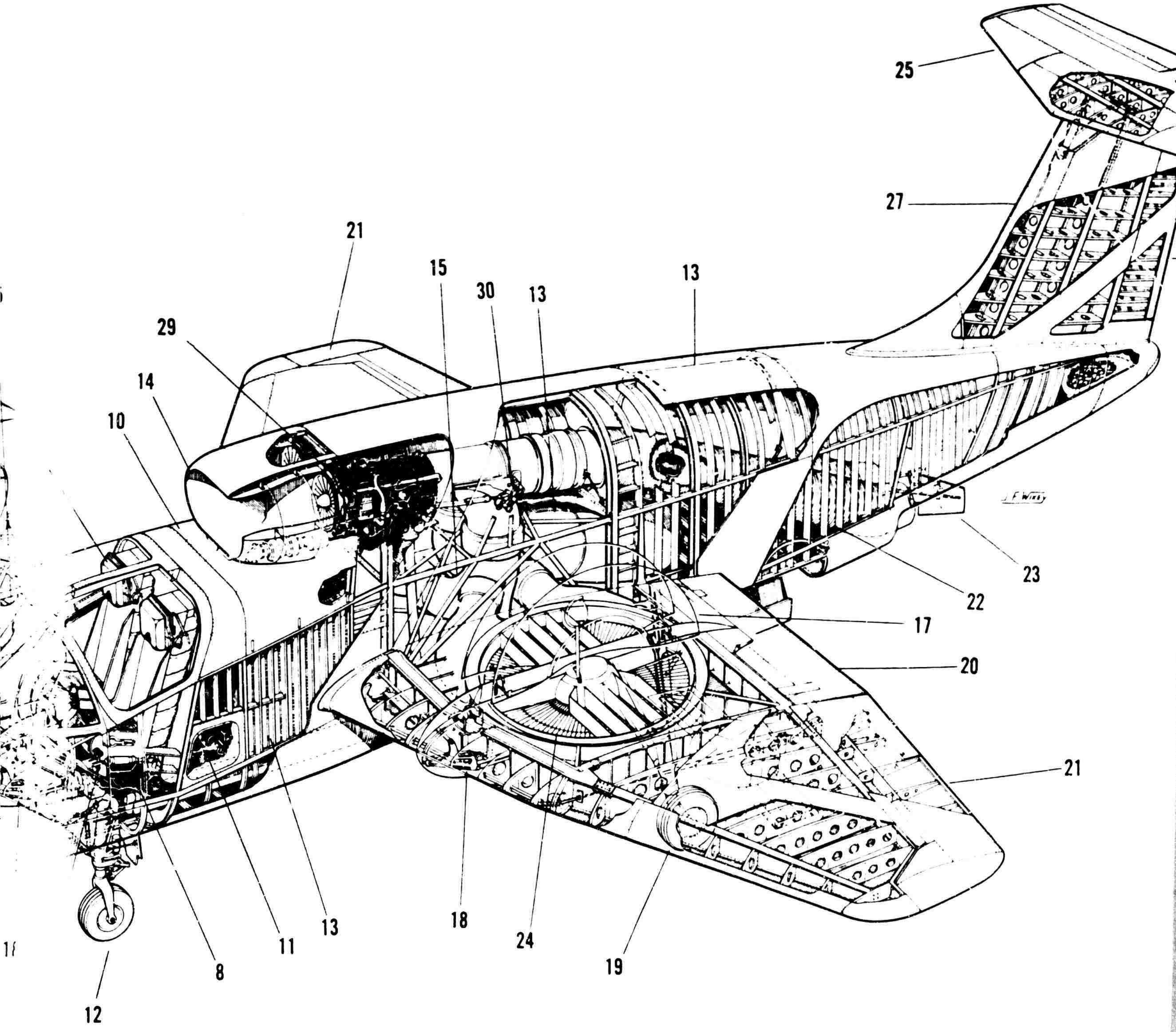

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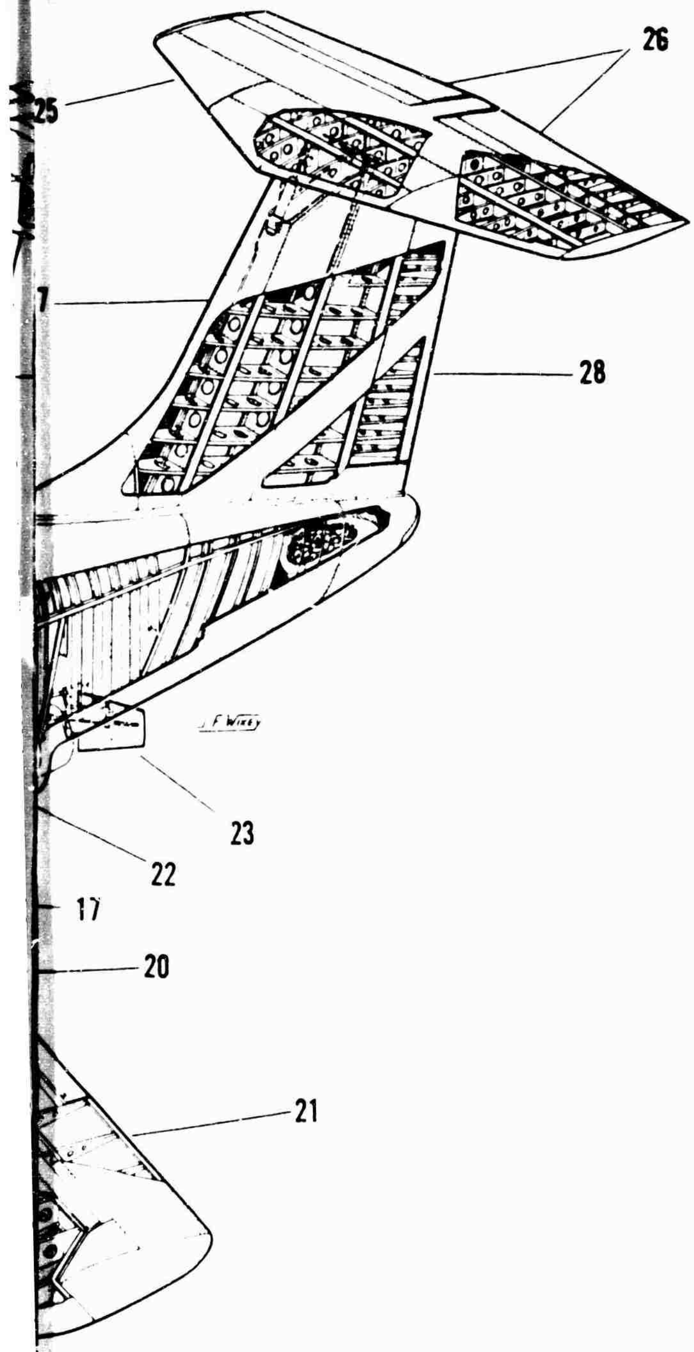


1. NOSE FAN INLET LOUVERS
2. GE-X376 PITCH CONTROL FAN
3. NOSE FAN THRUST REVERSERS AND NOSE FAIRINGS
4. ZERO-ZERO EJECTION SEAT
5. CONVENTIONAL CONTROL STICK
6. RUDDER PEDALS
7. THROTTLES
8. LIFT CONTROL STICK
9. INSTRUMENT PANEL
10. HYDRAULIC COMPARTMENT

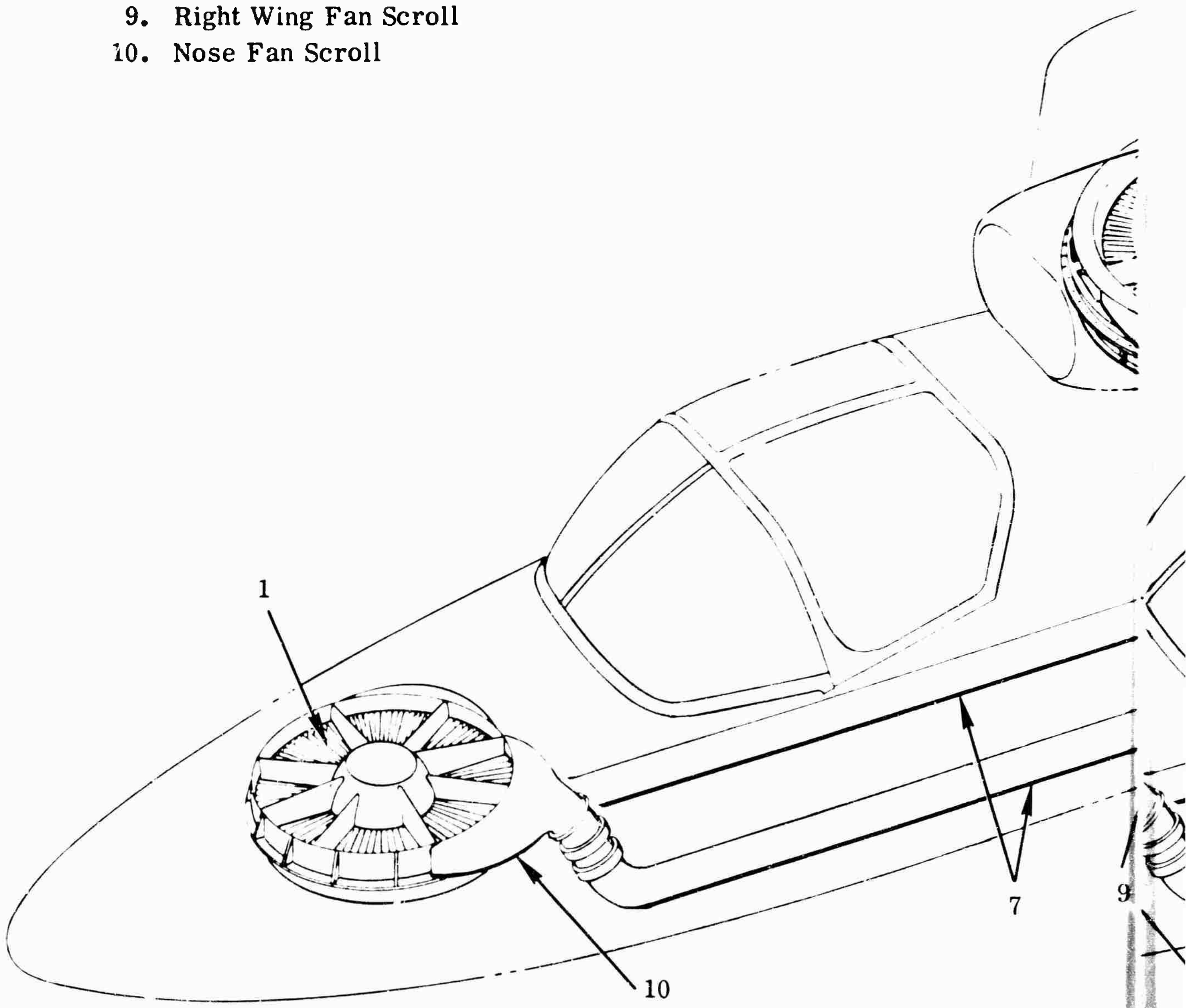
11. ELECTRICAL COMPARTMENT
12. NOSE GEAR
13. FUEL TANK
14. SINGLE SPLIT INTAKE
15. CROSSOVER DUCT
16. NOSE FAN SUPPLY DUCT
17. MAIN FAN CLOSURE
18. EXIT LOUVER ACTUATORS
19. TWO POSITION LANDING GEAR
20. FLAP

21. AILERON
22. ENGINE TAIL PIPES
23. THRUST SPOILERS
24. GE-X353-5 LIFT FAN
25. FULL MOVEABLE HORIZONTAL STABILIZER
26. ELEVATORS
27. VERTICAL FIN
28. RUDDER
29. GE J85 GAS GENERATOR
30. DIVERTER VALVE

Figure 2.1 General Arrangement XV-5A

c

1. Nose Fan
2. Gas Generator
3. Diverter Valve
4. Engine Tail Pipe
5. Wing Fan
6. Crossover Ducts
7. Nose Fan Supply Duct
8. Left Wing Fan Scroll
9. Right Wing Fan Scroll
10. Nose Fan Scroll



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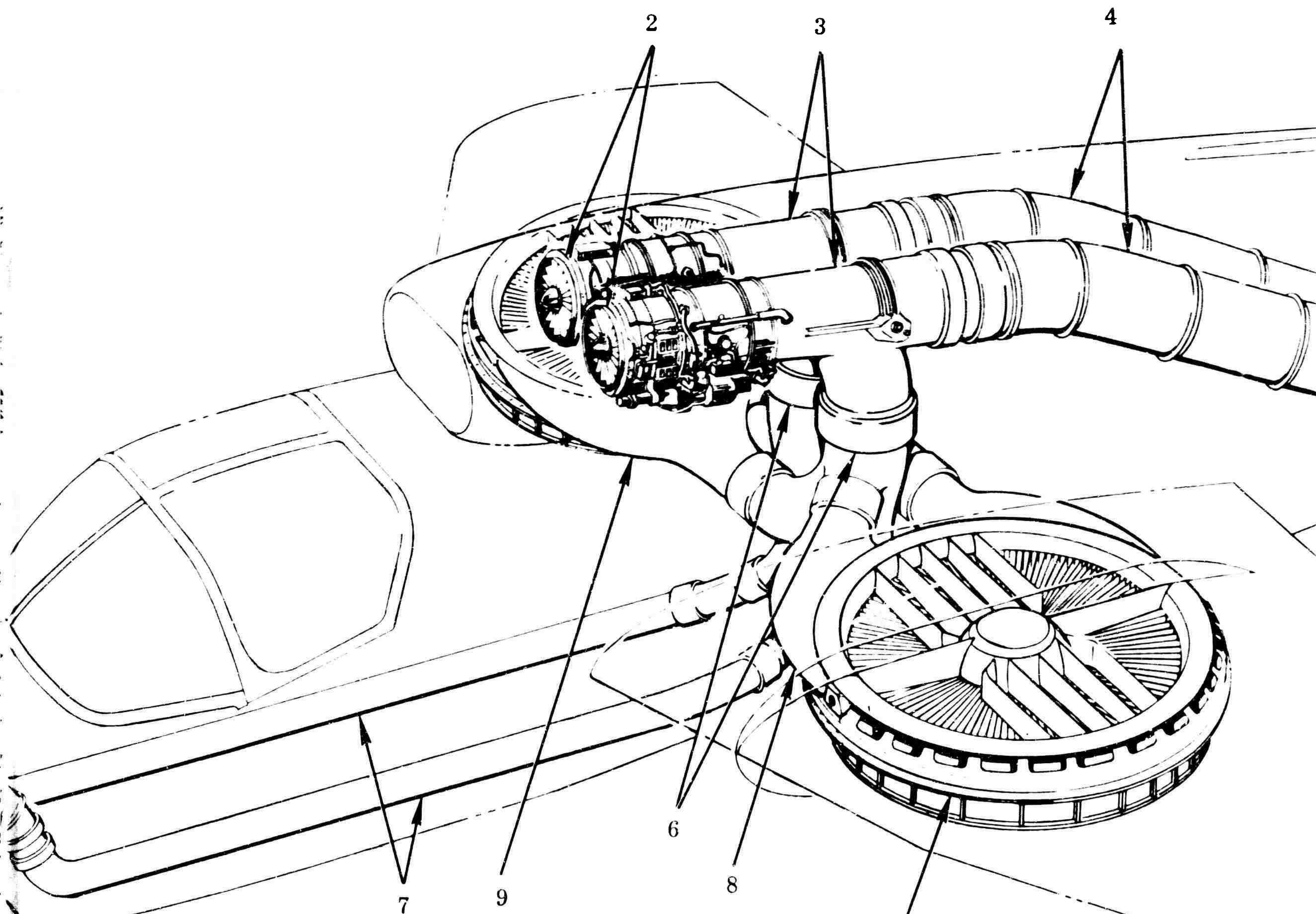

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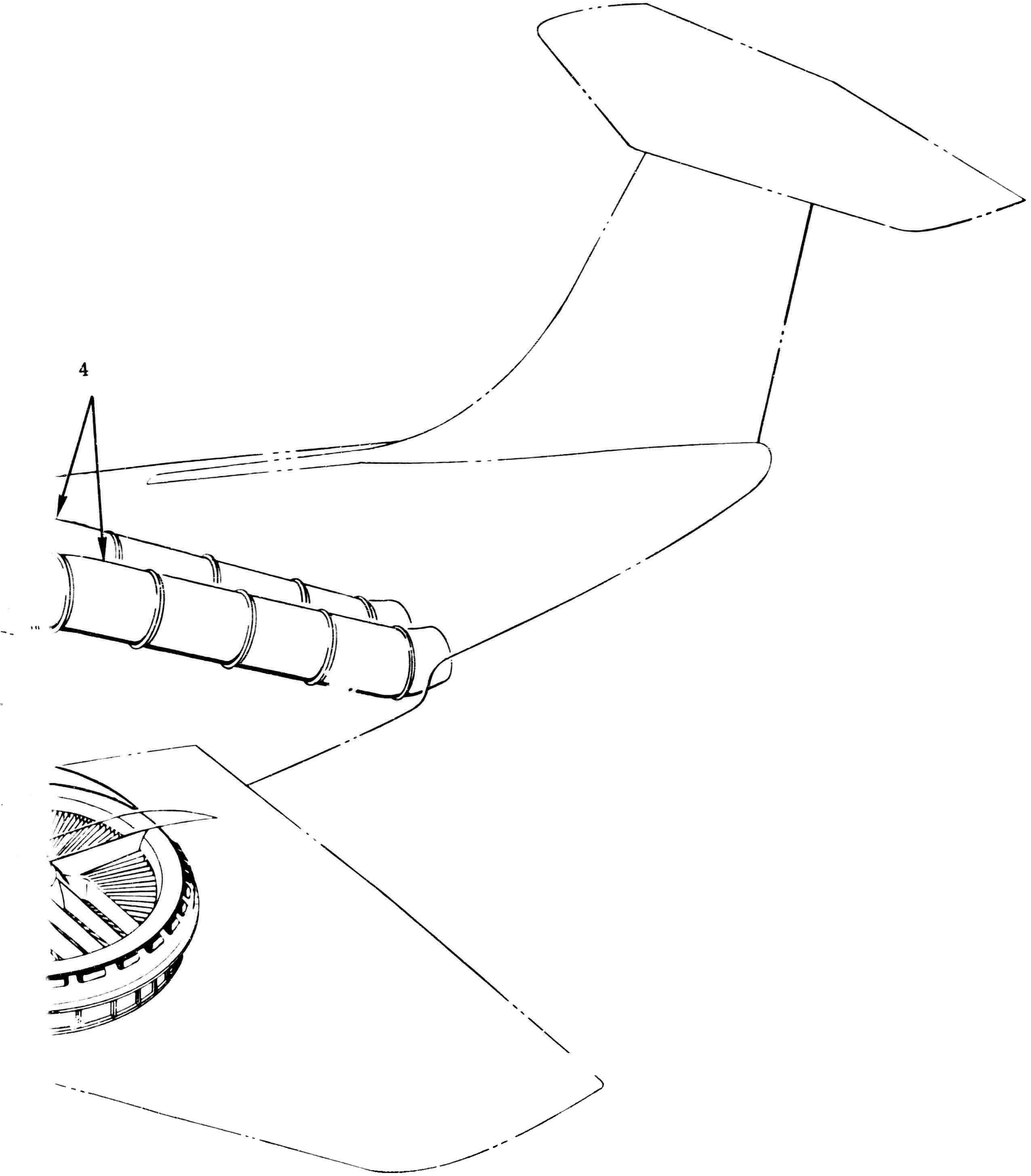

Figure 2.2 XV-5A Propulsion System

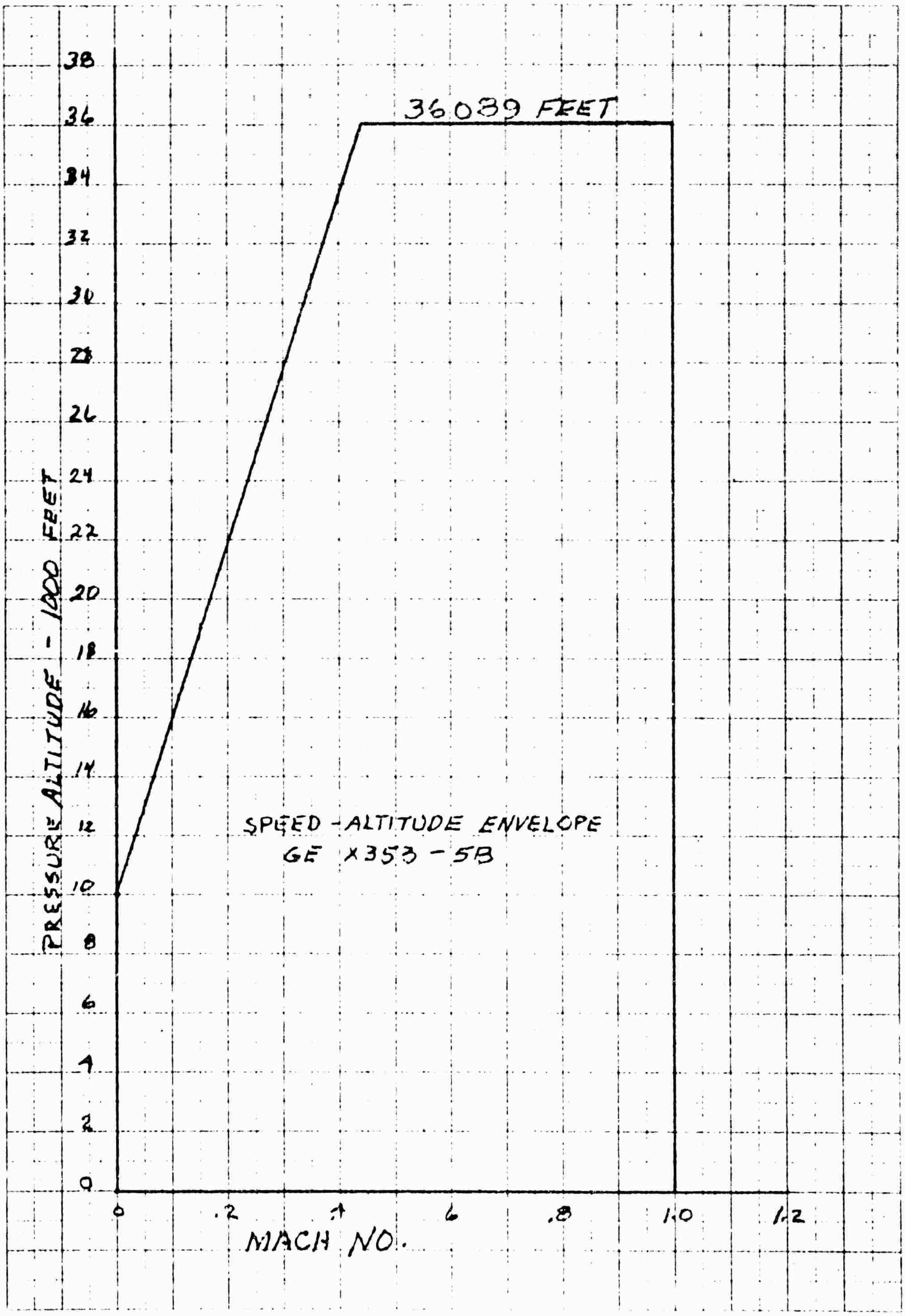

Figure 2.3 G. E. X353-5B Speed Altitude Envelope

3.0 INSTALLED PERFORMANCE LIFT FAN MODE OPERATION

3.1 GENERAL

Presentation of lift fan mode powerplant performance characteristics is limited to the static (zero speed) condition. These characteristics are discussed in the following Sections of this report. While References 1 through 6 of Section 7.0 may be utilized for the calculation of fan performance in the transition flight regime, the characteristics so derived to not represent installed performance. Instead, they represent portions of a total which are in effect inseparable from the whole and are only theoretically derivable and accountable as components. Thus, it is considered necessary to assume, as contributing to fan performance, some of the aerodynamic forces generated on the aircraft, such as on the wing leading edge immediately ahead of the fan, but not necessarily aerodynamic forces generated (or induced) on other areas of the airframe, such as wing undersurface, in order to arrive at the theorized contribution.

Inasmuch as installed fan-alone data (with or without bellmouth and exit louver system) was not available, it was decided early in the XV-5A program to work with the airframe and propulsion system in combination, in order to estimate the transition flight performance of the aircraft. With the assistance of General Electric Company, experimental data from tests of small scale wind tunnel models and a full scale model with actual G.E. tip-turbine lift-fan hardware, have been utilized exclusively for prediction of aircraft transition flight performance. The methods used in arriving at predicted aircraft characteristics; the data handling procedures, and the characteristics developed, are described in Reference 7. The techniques used in calculation of transition performance permit the determination of fan speeds from fan power absorption characteristics, as well as the airplane forces and moments as a function of power setting, airspeed, attitude with respect to flight path, and airplane configuration (louver angle, door position, etc.). In using the calculation procedures, all fan mode characteristics are referenced to installed static lift fan performance characteristics.

3.2 INSTALLED STATIC FAN MODE PERFORMANCE

Installed static propulsion system performance for fan mode operation has been provided directly by General Electric (See Reference 8 of Section 7.0 which is also reproduced in whole in this report in Section 7.2) for ARDC standard day and ANA bulletin 421 hot day conditions. These characteristics were developed using test results derived during the propulsion system flight worthiness test program performed at the General Electric Flight Propulsion Division facility. Previous to the receipt of this data, References 1 through 6 of Section 7.0 had been used to derive estimates of static propulsion system performance, and results obtained using the digital computer deck were assumed to express airplane characteristics applicable to operation out of ground effect. Subsequent review of model pressure data verified the early assumption that for the XV-5A configuration removed from ground proximity, fan slipstream induced pressures producing loads on the adjacent wing-fuselage undersurfaces were negligibly small. Also it was found that the propulsion system performance specifications reflected by the computer deck had been based on measured performance obtained from operation of the fan system in a wing panel where local effects had been automatically integrated into the data.

Reference 4 of Section 7.0 concerning the nose fan, specifies a 1500 pound lift capability for the design conditions of 13.08% bleed flow, and a 2,500 foot altitude ANA 421 Hot Day without correction for gas generator or nose fan inlet losses. Accounting for these effects and assuming a three percent loss in lift for nose fan inlet (due to the closure system), a maximum nose-fan lift of 1,360 pounds was calculated which was considered to be too low to meet hovering control requirements for design conditions. While the fan speed is somewhat higher than orignally specified, the pitch fan performance data of Reference 8 of Section 7.0 indicates lift slightly in excess of the originally requested 1,500 pounds at a bleed flow of only 12.3%. Somewhat improved performance is also indicated in Reference 8 of Section 7.0 for the main fans.

For convenience, Reference 8 of Section 7.0 is reproduced in whole in this report, and may be found in Section 7.2.

Installation factors considered and assumed in determining Static System Fan Mode Performance are as follows, (Reference 8 and Section 7.2):

1. Average pitch and main fan performance (based on full scale test of actual propulsion system).

2. Minimum J85 gas generator performance.

3. Main lift fan louver vector and stagger angles equal zero.

4. Pitch fan thrust reverser doors set at maximum lift position.

5. Inlet pressure recovery equal to 98.4%.

6. Power extraction from each gas generator equal to 58 H.P.

7. Compressor bleed air equal 0.022 lbs/sec.

8. Pitch fan and main fan inlet losses based on scale model tests.

9. No hot gas re-ingestion.

10. No ground effects.

11. Diverter valve, wing-fan cross-over and pitch-fan duct losses based on full scale tests.

12. Pitch-fan scroll area set for 12.3% bleed.

13. Wing-fan scroll area trimmed for exhaust gas temperature (EGT) for an ANA 421 hot day at conditions of 2500 feet altitude and 93.7°F ambient temperature.

14. Definition of 100% rpm:

    a. Gas generator 16,500 rpm

    b. Wing fan 2,640 rpm

    c. Pitch fan 4,074 rpm

3.3 CORRELATION OF INSTALLED STATIC FAN MODE PERFORMANCE

From the data noted in the References 1, 4, and 8 in Section 7.0, a convenient method of relating static lift fan and gas generator performance was developed. Presented in Figures 3.1 and 3.2 for wing lift fan and pitch control fan respectively, the correlation represents data of Reference 8 of Section 7.0. More important, however, are

the convenient insights that the correlation makes possible, regarding effects on fan mode performance of such factors as installation losses, altitude, ambient temperature, different air temperatures entering the gas generator, pitch and wing fans due to re-ingestion resulting from locally induced aircraft environments, % bleed to pitch fan, and average to guaranteed minimum gas generator performance.

3.4 RECOMMENDED FAN MODE PERFORMANCE STUDY

While the procedures used in determination of aircraft transition performance (where the airframe and propulsion system are treated as a unit) do not provide accountability of contributions, the techniques are nevertheless effective. Procedures for determination of net propulsion system contributions will unquestionably prove desirable, if not necessary, in the future. Thus, fan-mode powerplant contributions might be given as a function of fan-inlet inflow characteristics, including velocity and/or pressure distributions. Conditions at the exit should also be defined in terms of back pressure, to cover effects of exit louvers, wing local pressure distributions, etc. The determination of the fan inflow and outflow characteristics, which would be required to utilize these performance calculation procedures, would however, be an extremely difficult task. Nevertheless, it is recommended that a study be undertaken, and research be conducted, to develop this or some other method of accurately defining propulsion system contributions to aircraft performance.

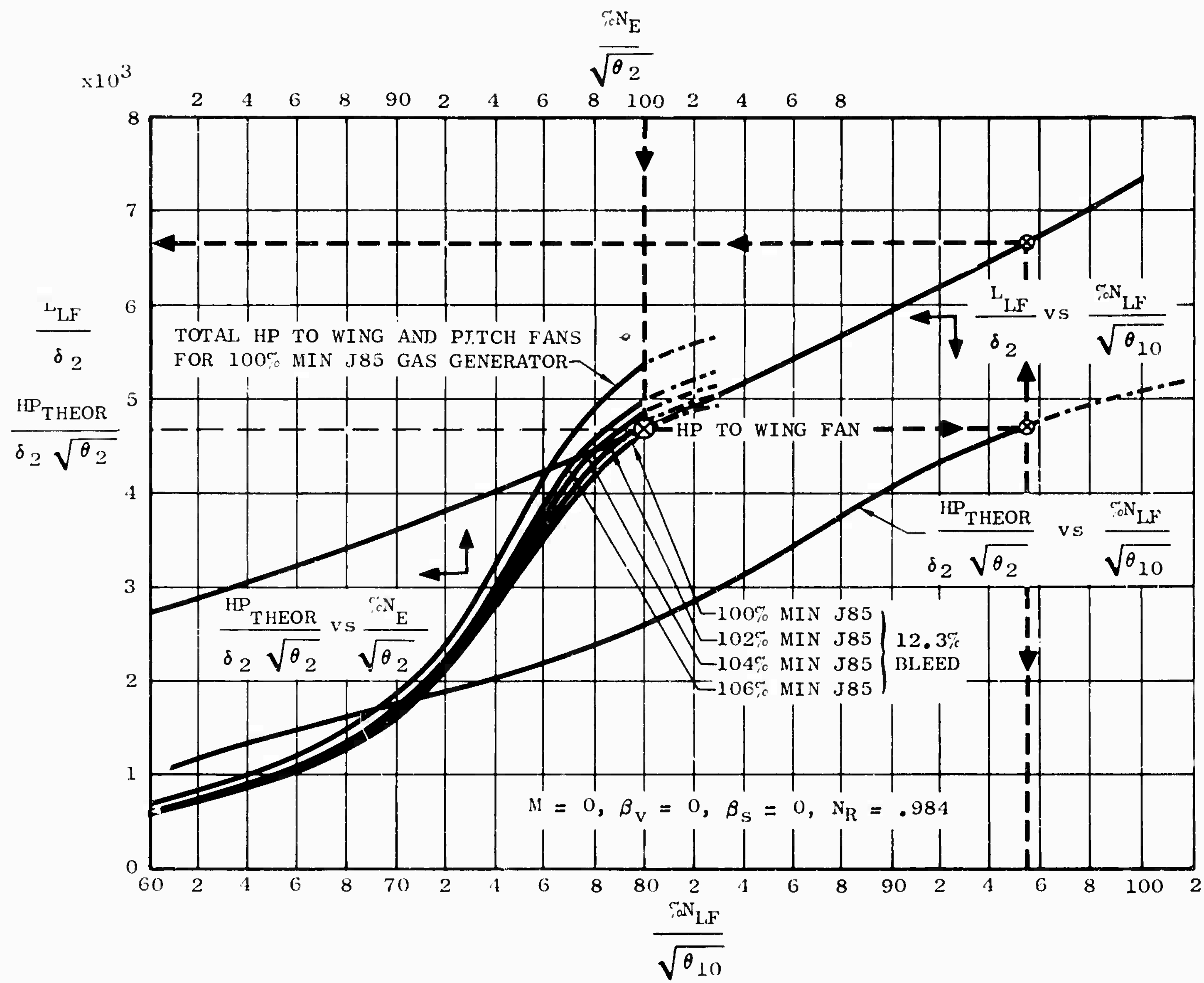

Figure 3.1 Estimated XV-5A Installed Power Plant Static Performance - Wing Lift Fan

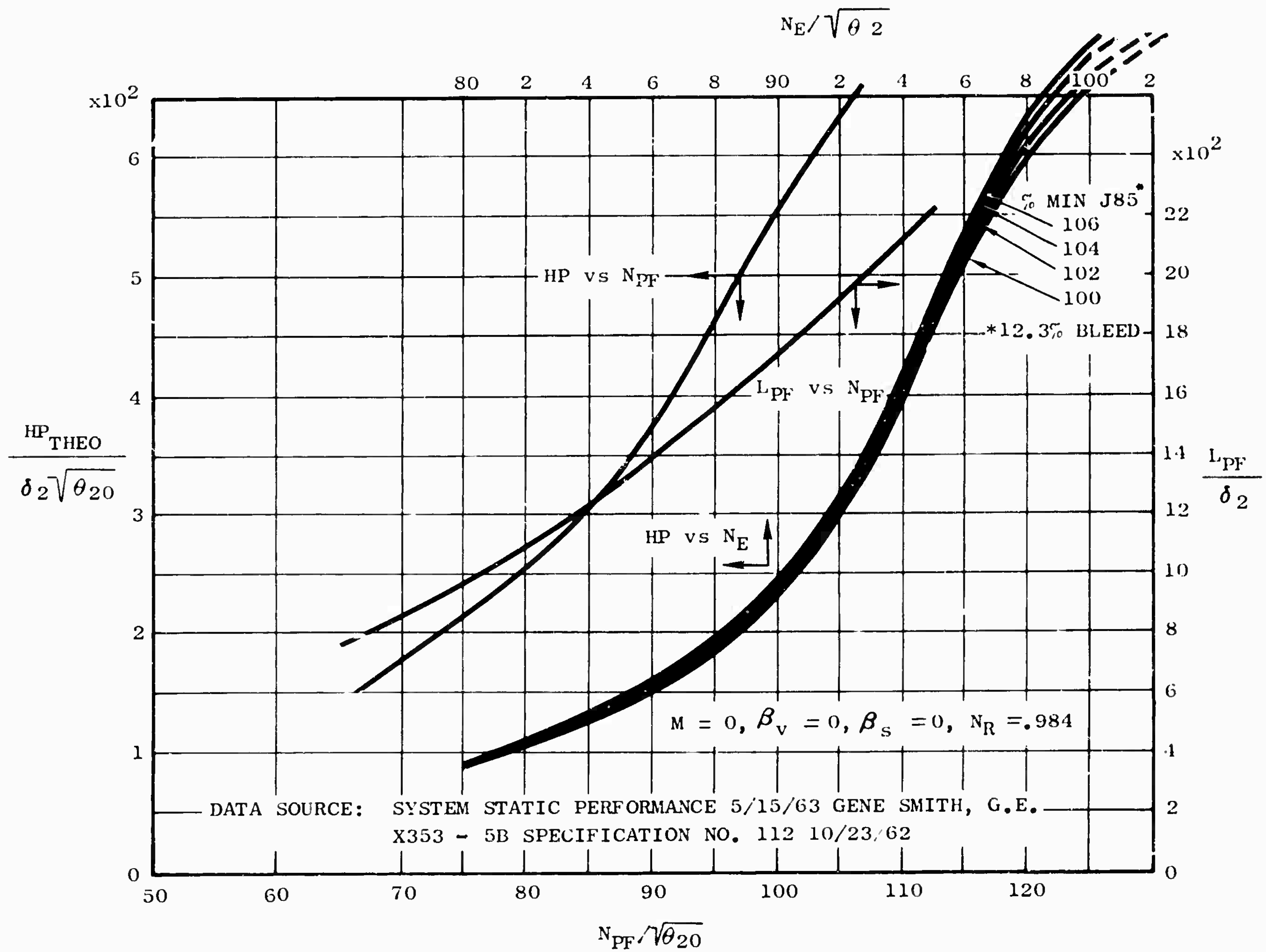

Figure 3.2 Estimated XV-5A Installed Power Plant Static Performance - Pitch Control Fan

4.0 INSTALLED PERFORMANCE - TURBOJET MODE OPERATION

4.1 GENERAL

Turbojet mode performance of the GE X353-5B power plant installed in the XV-5A aircraft is based on the following sources of data:

- Guaranteed minimum performance of the GE X353-5B power plant is provided by References 1, 9 and 10 of Section 7.0, which include effects of diverter-valve leakage and pressure loss.

- Inlet pressure recovery factors used are presented in Section 4.6 based on the data of Reference 11 for one and two engine operation.

- External incremental drag data were derived from Reference 12 of Section 7.0 and verified by wind tunnel tests. The drag coefficients are presented in terms of flight speed and mass flow ratio, and are referenced to the minimum inlet area.

- Ejector performance affecting gross thrust was derived from data described in Reference 13 of Section 7.0. Ejector performance factors are presented in terms of tailpipe to ambient pressure ratio, and secondary air to ambient pressure ratio. The former is obtained from the computer deck indicated in Reference 10 of Section 7.0, and the latter from cooling system performance of Reference 14.

The method of estimating installed performance for the GE X353-5B power plant in turbojet mode involved the following steps:

Step 1

The area of the turbojet exhaust nozzle was sized by assuming a matrix of inlet recovery factors and tailpipe nozzle areas, and obtaining installed performance via the computer deck indicated in Reference 10 of Section 7.0. From the output, computed inlet recoveries were plotted in terms of

assumed values to obtain the condition of equality for each area. The harness exhaust gas temperature (HRGT) also was plotted vs assumed inlet recovery factor, and the proper value selected for each area. Then the HRGT values were plotted vs assumed nozzle area and the correct area of 111.54 square inches was read at an HRGT value of 1710°R.

Later it was determined that tailpipe temperature (T8) should have been used, rather than HRGT to size the nozzle area. Re-sizing the area, as above, showed a requirement of 110.96 square inches. Spot-checks of installed performance (See Table 4.1) showed the propulsion system drag was unaffected by the area change and that the gross thrust was increased by approximately 1%. The increase in net thrust therefore is dependent upon the magnitude of ram drag. Generally the increase is about 1% except at high ram drag conditions where the increase may amount to as much as 3%.

The XV-5A aircraft tailpipe has an exit area adjustment using tabs which may be used to vary tailpipe area over a range of 111.03 to 114.05 square inches.

Step 2

Following nozzle area sizing, an input data matrix was prepared slightly exceeding the speed-altitude envelope of Figure 2.2 and a range of power plant operation from 75 to 100% RPM for both standard and hot day conditions. Again it was necessary to assume a range of inlet recovery factors as a primary variable. This input data matrix was based on compressor face inlet conditions of total temperature and pressure.

Step 3

The data of Step 2 was combined with the performance computer deck (see Reference 10) to obtain an initial output matrix of installed performance data in preparation for final calculation of installed performance.

Step 4

To eliminate extensive manual manipulation of the data of Step 3, a special digital computer routine, designed to accept the data of Step 3 as an input, was used to iterate calculated vs assumed inlet recovery factors; to interpolate performance data at the correct inlet recovery; and to calculate additional performance factors such as external drag, windmilling drag for engine-out conditions, and ejector contributions

TABLE 4.1

Tailpipe Area Effect on Installed Performance of G. E. X353-5B Power Plant-Turbojet Mode
2 Engine Operation, ANA Bulletin 421 Hot Day

			CASE I-TAILPIPE AREA = 110.96 in.^{2}				CASE II-TAILPIPE AREA = 111.54 in.^{2}				PERFORMANCE FACTOR (CASE I/CASE II)			
Altitude	% RPM	Mach No.	Total Gross Thrust	Total Drag	Total Net Thrust	Total Fuel Flow	Total Gross Thrust	Total Drag	Total Net Thrust	Total Fuel Flow	Total Gross Thrust	Total Drag	Total Net Thrust	Total Fuel Flow
0	100	0	4288	0	4288	4674	4245	0	4245	4616	1.010	1.0	1.010	1.026
		.4	4896	1226	3670	4890	4851	1226	3625	4827	1.009	1.0	1.012	1.013
		.6	5431	2076	3355	4977	5377	2076	3300	4905	1.010	1.0	1.017	1.015
	95	.0	3036	0	3036	3420	3009	0	3009	3388	1.009	1.0	1.009	1.009
		.4	3279	1096	2183	3356	3247	1096	2151	3318	1.010	1.0	1.015	1.011
		.6	3591	1893	1698	3324	3555	1893	1662	3282	1.010	1.0	1.022	1.013
	85	0	1550	0	1550	2125	1539	0	1539	2116	1.007	1.0	1.007	1.003
		.4	1800	934	866	2158	1788	934	854	2147	1.007	1.0	1.014	1.005
		.6	2018	1658	360	2061	2004	1658	346	2049	1.007	1.0	1.040	1.006
5,000	100	0	3903	0	3903	4226	3865	0	3865	4174	1.010	1.0	1.010	1.012
		.4	4506	1064	3442	4493	4466	1064	3402	4438	1.009	1.0	1.011	1.012
		.6	5151	1800	3351	4663	5107	1800	3307	4623	1.008	1.0	1.013	1.013
	95	.0	2906	0	2906	3142	2874	0	2874	3103	1.011	1.0	1.011	1.013
		.4	3200	960	2240	3176	3148	960	2187	3111	1.017	1.0	1.024	1.021
		.6	3420	1638	1782	3104	3380	1638	1742	3056	1.012	1.0	1.023	1.016
	85	.0	1449	0	1449	1961	1437	0	1437	1950	1.008	1.0	1.008	1.006
		.4	1570	804	766	1787	1559	804	755	1777	1.007	1.0	1.016	1.006
		.6	1827	1427	400	1836	1814	1427	387	1824	1.007	1.0	1.034	1.007
10,000	100	0	3488	0	3488	3755	3454	0	3454	3707	1.010	1.0	1.010	1.013
		.4	4080	915	3165	4057	4038	915	3123	4000	1.010	1.0	1.013	1.014
		.6	4705	1549	3156	4295	4667	1549	3118	4242	1.008	1.0	1.012	1.012
	95	.0	2735	0	2735	2889	2697	0	2697	2842	1.014	1.0	1.014	1.017
		.4	3079	835	2244	2983	3045	835	2210	2939	1.011	1.0	1.015	1.015
		.6	3322	1419	1903	2976	3287	1419	1868	2932	1.011	1.0	1.019	1.015
	85	.0	1322	0	1322	1701	1312	0	1312	1691	1.008	1.0	1.008	1.006
		.4	1466	687	779	1663	1455	687	766	1654	1.008	1.0	1.014	1.005
		.6	1604	1219	385	1530	1592	1219	373	1520	1.008	1.0	1.032	1.007

to gross thrust. Step 4 provides installed performance for the GE X353-5B powerplant operating in the turbojet mode. Data presented include gross thrust, fuel flow, total drag and net thrust, exclusive of cooling system drag, which is treated separately in this report. Parameters include one and two engine operation for standard and hot day conditions for the required range of flight speeds, altitudes, and J85-GE-5 gas generator speeds from 75% to 100%.

4.2 INSTALLED PERFORMANCE

The following factors affecting installed performance of the GE X353-5B power plant installed in the XV-5A aircraft were held constant:

Power Extraction	40 Horsepower
Compressor Bleed Air	0.022 pounds/sec.
Exhaust Duct Pressure Loss	3.1%
Nozzle Area	111.54 square inches

Estimated installed performance, including gross thrust, propulsion system drag, net thrust and fuel flow, is presented in Figures 4.1 through 4.104. The gross thrust vector is at an angle of 6.5° with respect to the aircraft axis. In providing net thrust, the gross thrust vector was not resolved because the ultimate error involved is less than 20 pounds of thrust for the total aircraft, and this occurs at the maximum speed minimum altitude condition. Actually, the ultimate error of not resolving the gross thrust vector is directly related to the ram drag component of gross thrust, since in estimating aerodynamic performance, it is assumed the net thrust vector is at an angle of 6.5° and is resolved in computing aircraft forces. Component performance data used in estimating final installed performance are summarized in Tables 7.3.1 through 7.3.6 as indicated in Step 1 of the calculation procedure. The data of Figures 4.1 through 4.104 should be increased by the factors of Table 4.1. Cooling system drag must be accounted for in determining thrust available for XV-5A aerodynamic performance estimates.

4.3 IDLE POWER OPERATION

When estimating installed performance data at idle power, the output from the IBM Customer Performance deck (see Reference 10 and step 3, Paragraph 4.1) showed that input variable limits had been exceeded.

A review of horsepower extraction values indicated limits were not exceeded for 10 HP or less, but were exceeded for 15 HP or more.

A review of horsepower extraction requirements for pre-flight checkout showed an average value of 19.9 HP and a peak value of 38.9 HP for 8 to 10 seconds. Subsequent discussions with General Electric Company developed the information of Figure 4.105, and resulted in a lifting of the horsepower extraction limits to normal J85-5 limits. The data of Figure 4.105 reveals that severe power extraction limits were imposed by both the GE X353-5B Model Specification (Reference 1) and the Customer Computer deck, Reference 10 of Section 7.0.

Even with the relaxed limit, however, an engine rpm of 52% is required to meet the peak demands of 38.9 HP. This particular value is in the undesirable speed range of 50 to 58% rpm for steady-state operation (See Reference 15). Therefore, pre-flight system checkout will require engine speeds of 60% rpm or greater. This causes no hardship because other reasons outlined below also dictate idle power settings greater than 60% rpm. Figure 4.106 shows that the minimum exhaust gas temperature (EGT) occurs between 70 and 80 rpm, dependent to some extent upon ambient temperature. Study of Figure 4.106 also reveals a tendency for EGT to exceed the allowable limit of 1710°R (1250°F or 677°C) near idle power. This tendency was the subject of a more thorough evaluation for sea level static conditions in order to explore effects of possible re-ingestion in fan mode operation. Results of this study are summarized in Figure 4.107, which presents exhaust gas temperature in terms of ambient temperature and % rpm. In a similar study, summarized in Figure 4.108, the effects of horsepower extraction on exhaust gas temperature were evaluated. In both studies, it was found that increasing ambient temperature or horsepower extraction caused a rapid increase in exhaust gas temperature when the gas generators were operating near idle speeds (47 to 50% rpm). Considering the likelihood of re-ingestion when operating the XV-5A in the fan mode, it is evident that engine over-temperatures are possible at engine speeds below 65% rpm, but not likely at speeds above this value, except possibly at 100% rpm. Another factor more or less directly related to engine speed is the cooling system performance. Since the cooling air blowers are directly connected to the engine through the gear box assembly, the higher the gas generator speed, the higher the cooling system performance. Thus, it is apparent that engine speeds approaching 75% are desirable for providing minimum EGT; for maximizing cooling system performance, and generally for promoting minimum aircraft temperatures. Countering these effects are the increasing braking power requirements with increasing rpm resulting from the increasing thrust, as shown in Figure 4.109.

4.4 EJECTOR PERFORMANCE

The XV-5A ejector configuration presented in Figure 4.110 is nearly identical to one of the full scale ejectors tested (Reference 13 of Section 7.0) from which its estimated performance was derived. Only that ejector performance affecting gross thrust is considered in this report. The presence of the ejector may increase or decrease gross thrust, depending upon the conditions of shroud air (secondary air) and turbojet exhaust (primary air) at the ejector. The ratio of thrust, with and without the ejector, is correlated in terms of two pressure ratios. These are secondary air to ambient pressure ratio, and primary air to ambient pressure ratio. The primary air to ambient pressure is a function of aircraft and gas generator operating conditions as they are affected by installation factors. It is derived from the initial computer deck output, (See Step 3, Section 4.1). The secondary air to ambient pressure is dependent upon the aircraft cooling system performance, which in turn is a function of aircraft and gas generator operating conditions. It is determined only when the cooling system shroud air flow rates and pressures match the ejector pumping capabilities. Analysis indicated this ratio was independent of ambient temperature and altitude for a given Mach number and gas generator rpm. Spot checks of conditions for cooling system balance at extremes of altitude and ambient temperature validated this conclusion and led to the data of Table 7.6 of this report.

Details of ejector performance are shown in Reference 14 of Section 7.0. Data used for estimating gross thrust are summarized in Tables 7.3.5 and 7.3.6. They form a part of the computer routine of Step 4, Section 4.1.

4.5 THRUST SPOILER PERFORMANCE

The thrust spoiler function is to permit high power levels when approaching VTOL conversion speeds from conventional flight conditions, while at the same time, reducing the net axial thrust. In the retracted position, the thrust spoilers form a fairing centered between and slightly aft of the turbojet nozzles. In the extended position, the slightly concave and tapered blades intercept the turbojet exhaust gases to deflect them symmetrically about the aircraft axis (see Item 17, Figure 2.1).

The reduction in axial thrust is determined by the method of approach outlined below and as shown in Figures 4.111, 4.112, and 4.113 based on References 16 and 17 of Section 7.0. The main problem is to estimate the effective jet deflection in terms of percent of thrust

spoiler deflection. Having an estimate of the jet deflection, the net axial thrust is determined simply by forming a product of the cosine of the deflected jet angle and the gross thrust vector from which is subtracted the propulsion system drag and the cooling air drag. Comparing the resulting net axial thrust with that of the undeflected jet yields the fraction or percent reduction at the given aircraft operating condition and thrust spoiler deflection.

Thrust spoiler angles expressed in terms of percent of thrust spoiler deflection are presented in Figure 4.111. In the retracted position (0% deflection) the thrust spoiler forms a relative angle of -7° with respect to the undeflected jet centerline. In the maximum extended position (100% deflection, 81° of travel) the relative angle is 74°. Because the base of the thrust spoiler is offset approximately 1 inch laterally from the projected tailpipe nozzle, the thrust spoiler must move through an angle of 10.8° from its retracted position (13.3% deflection) before it begins to enter the jet. Thus, the effective deflection angle of the thrust spoiler is the actual angular deflection less 10.8°. In a given effective thrust spoiler position, the jet actually is deflected somewhat less as shown in Figure 4.111.

Knowing the effective jet deflection, the estimated reduction in axial net thrust is obtained from data as shown in Figure 4.112 for a particular set of aircraft operating conditions. Figure 4.113 summarizes the range of estimated thrust spoiler performance in terms of percent of thrust spoiler deflection. Also shown is a derived data point based on XV-5A full-scale model thrust-spoiler tests at the NASA-Ames Laboratory. The substantial agreement with Curve A (Figure 4.113) lends support to the effective thrust spoiler angle concept.

Some concern existed over the possibility of ejector reverse flow at high thrust spoiler deflections. Although no pertinent factual data were uncovered in the literature, recent static ground tests of the XV-5A indicate reverse flow in the ejector will not occur until after 80% deflection. Since XV-5A thrust reductions beyond 64% are not required, no reverse flow problems are expected in the use of the spoilers.

4.6 INLET PERFORMANCE

As shown in Figure 2.1, the inlet is located at the top of the aircraft slightly aft of the cockpit canopy and slightly forward of the leading edge of the wings. In this position, inlet performance is affected somewhat by air flow over the canopy, and may be affected by upwash from the

interaction of pitch control and lift fan downwash patterns during lift mode operation in ground effect. Tending to offset canopy effects are the two sources of boundary layer removal: the first at the rear of the canopy closure which is unsealed and serves as a primary source of air to the aircraft cooling system; the second at the boundary layer bleed duct centered directly beneath the inlet primarily during turbojet mode operation. A total of thirteen simulated aircraft operating configurations (see Table 4.2) were investigated during the inlet model wind tunnel test program conducted at David Taylor Model Basin (DTMB) (Reference 11). Model configurations included three inlets (24E oval, 30E oval, and 30E dual), two inlet splitters (long and short), and two canopies (basic and cut-down). Operating configurations included one and two engines operating, and boundary layer duct open or closed. Suction was used to induce the required inlet airflows during the low speed portion of the test program. From the results of the wind tunnel tests, the aircraft inlet configuration selected was a minor modification of the 30E oval inlet with the long splitter, and the basic canopy. Only internal lines of the 30E configuration were modified. The upper outboard duct quadrant, beginning immediately aft of the throat, was filled in to reduce the diffusion angle of this duct section, and thus improve the pressure distribution shown at the compressor face for the 30E duct configuration. In addition, to improve single-engine inlet performance, a duct splitter with an elliptical leading edge was substituted for the original splitter which had a sharp leading edge. This selection offered the best over-all inlet performance for the operating range of the XV-5A aircraft with specific consideration of its VTOL and Mach = 0.7 cruise speed capabilities. Table 4.2 summarizes the comparisons of inlet recovery and distortion performance parameters (NRI, K and L) for the 30E inlet with the other inlet systems evaluated by Reference 11 of Section 7.0. The actual inlet as fabricated has a capture to throat area ratio of 1.29 and should be designated as a 29E inlet. The lip section contours for the inlet model (and the aircraft) were based on the elliptical lip coordinates of NACA TN3170. The external lines were based on NACA Series 1 coordinates. Nacelle lines were based on the 24E inlet starting at Station 170 and ending at Station 203.25 where it met the aft fuselage line. To hold this point for the 30E inlet, the coordinate line ran only 16.5 inches from which point (Station 186.8) loft lines were faired to the fuselage lines at Station 203.25.

Use of the modified long splitter reduced the inlet minimum area from 183.9 to 182.3 square inches, and moved its location aft approximately four inches as shown by the area-station profile of Figure 4.114. Although Schlieren studies showed that the cut-down canopy improved high

TABLE 4.2

Ryan XV-5A 1/5 Scale Inlet Model
Configuration Notation and Configurations Tested

Configuration Notation

C 1	Basic Canopy
C 2	Cut Down Canopy
I 0	24E Oval Inlet
I 1	30E Oval Inlet
I 2	30E Dual Inlet
S 0	Short Splitter Plate
S 1	Long Splitter Plate
S 2	Dual Inlet Splitter Plate
B 0	Boundary Layer Duct Plug Closed
B 1	Boundary Layer Duct Plug Open
E 1	Single Engine Operation
E 2	Two Engine Operation

Configuration Tested

C	I	S	B	E	Low Speed Wind Tunnel*	High Speed Wind Tunnel*
1	1	1	1	2	X	X
1	0	1	1	2	X	X
1	1	0	1	2	X	-
1	1	0	0	2	X	-
1	0	0	1	2	X	-
1	0	0	0	2	X	-
1	2	2	1	2	X	X
1	2	2	0	2	X	-
2	1	1	1	2	X	-
1	1	1	1	1	X	-
1	0	1	1	1	X	X
1	0	0	1	1	X	-
1	2	2	1	1	X	X

* X Indicates configuration tested

\- Indicates configuration not tested

speed characteristics, cockpit and aircraft considerations favored the basic canopy; this was particularly true, since it permitted adequate aircraft performance.

Inlet performance data used in determining installed power plant performance were prepared for trimmed flight conditions (See Figure 4.115) based on the wind tunnel inlet model data of Reference 11 with no allowance for inlet improvements discussed previously. They are presented in Figures 4.116 and 4.117 for two and one engine operating conditions respectively. Also shown on Figure 4.116 are typical values of mass flow ratio for a range of XV-5A aircraft operating conditions; from which it is evident the inlet shows excellent performance over its required operating range. Cross plots of these data led to the inlet recovery factors of Table 7.2 and 7.3 which were used to calculate installed performance.

Aircraft attitudes at other than trimmed flight conditions are considered transitory in nature. For these conditions primary concern is to minimize compressor face pressure distortion. Typical distortion parameters for a range aircraft attitudes are presented in Figures 4.118 to 4.124, which show inlet performance to be relatively insensitive to aircraft attitudes likely to be encountered. Data of Figures 4.121 and 4.122 are for the 24E oval inlet since the 30E oval inlet model was not tested at $M = 0.4$ and $M = 0.6$. A comparison of the 24E and 30E oval inlets at $M = 0.7$ is provided by Figures 4.123 and 4.124. The effect of simulated cross winds or side slip angles of ±30° is presented in Figures 4.125 to 4.128. These figures show greater distortion is experienced by the upwind engine and that the distortion increases with increased engine power as represented by the mass flow ratio m/m^*.

Estimated external drag coefficients used are presented in Figure 4.129. They were derived from Reference 12 of Section 7.0. The spot check, derived from available conventional wind tunnel model data, showed excellent agreement with the estimated data.

TABLE 4.3

XV-5A Inlet Model Configuration
Performance Comparison Chart

NOTE: (>) = greater than, (<) = less than, (=) = essentially equal.
Example: NRI > means 11112 greater than 11012, etc.

MACH NO.	30E OVAL VS CONFIGURATION C I S B E	INLET	C I S B E	NRI	K	L
0	1 1 1 1 2 vs	30E Oval	1 1 0 1 2	=	=	=
			1 1 0 0 2	=	=	=
		24E Oval	1 0 1 1 2	>	<	=
			1 0 0 1 2	>	<	>
			1 0 0 0 2	>	<	=
		24E Dual	1 2 2 1 2	=	=	>
			1 2 2 0 2	=	=	>
.15	1 1 1 1 2 vs	24E Dual	1 2 2 1 2	=	=	>
		30E Oval	1 1 0 1 2	=	=	=
			1 1 0 0 2	=	=	=
			2 1 1 1 2	=	=	>
		24E Oval	1 0 1 1 2	>	<	>
			1 0 0 1 2	>	<	>
		30E Oval	1 1 0 1 2	=	=	=
			1 1 1 1 1	>	<	>
	1 1 1 1 1 vs	24E Oval	1 0 1 1 1	>	=	=
			1 0 0 1 1	<	>	=
		24E Dual	1 2 2 1 1	<	<	>
.40	1 0 1 1 2 vs	24E Oval	1 0 1 1 1	=	=	=
		24E Dual	1 2 2 1 2	=	=	=
.6	1 0 1 1 2 vs	24E Dual	1 2 2 1 2	=	=	=
		24E Oval	1 0 1 1 1	≅	≅	≅
.7	1 1 1 1 2 vs	24E Oval	1 0 1 1 2	>	<	>
		24E Dual	1 2 2 1 2	>	<	<
			1 2 2 1 1	>	<	<
		24E Oval	1 0 1 1 1	>	<	=
.8	1 1 1 1 2 vs	24E Oval	1 0 1 1 2	>	<	>
		24E Dual	1 2 2 1 2	>	<	<

NRI - Total Pressure Recovery
K - Total Pressure Distortion
L - Static Pressure Distortion
See Table 4.2 for Configuration Symbol Definition

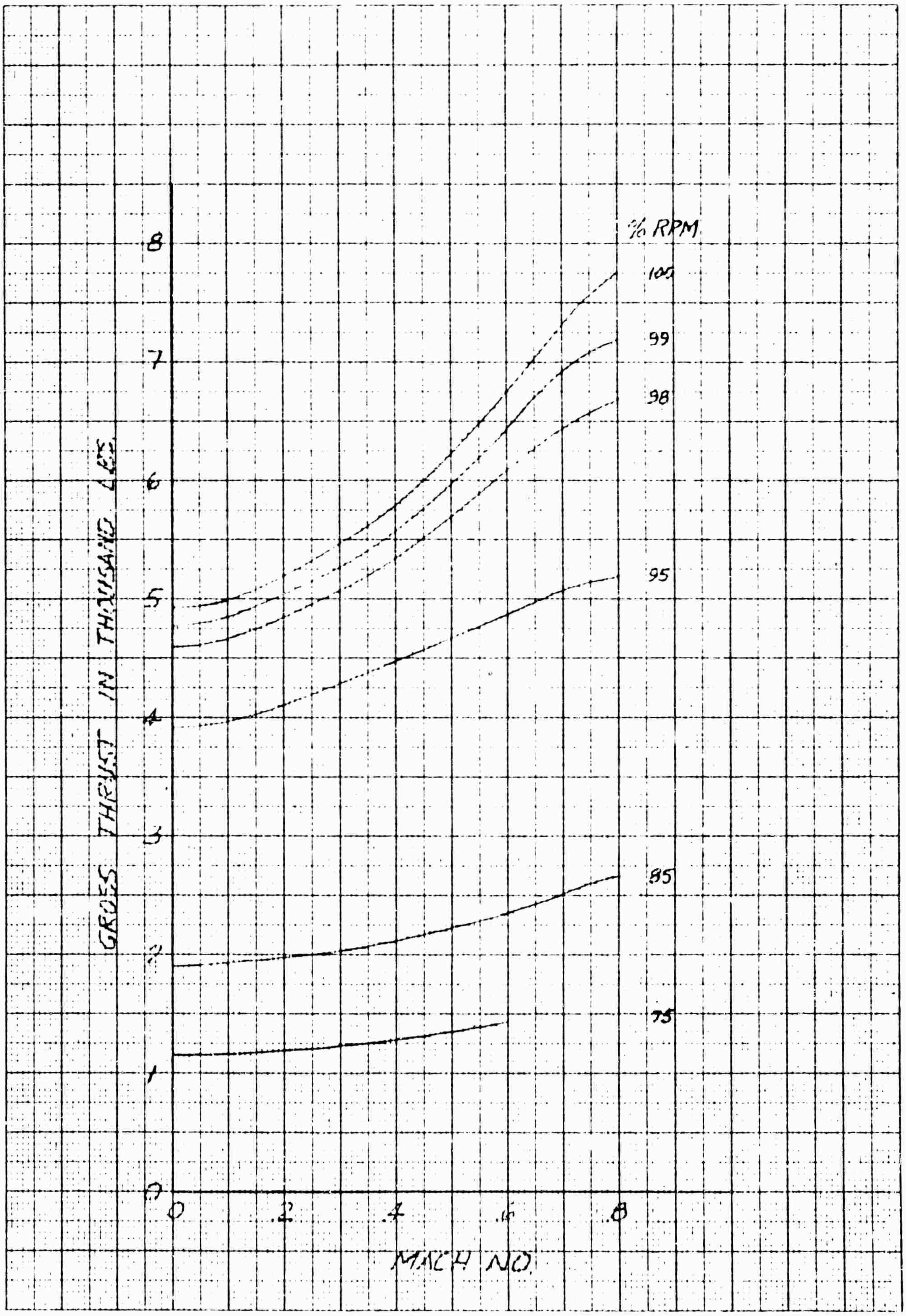

Figure 4.1 Gross Thrust vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Standard Day

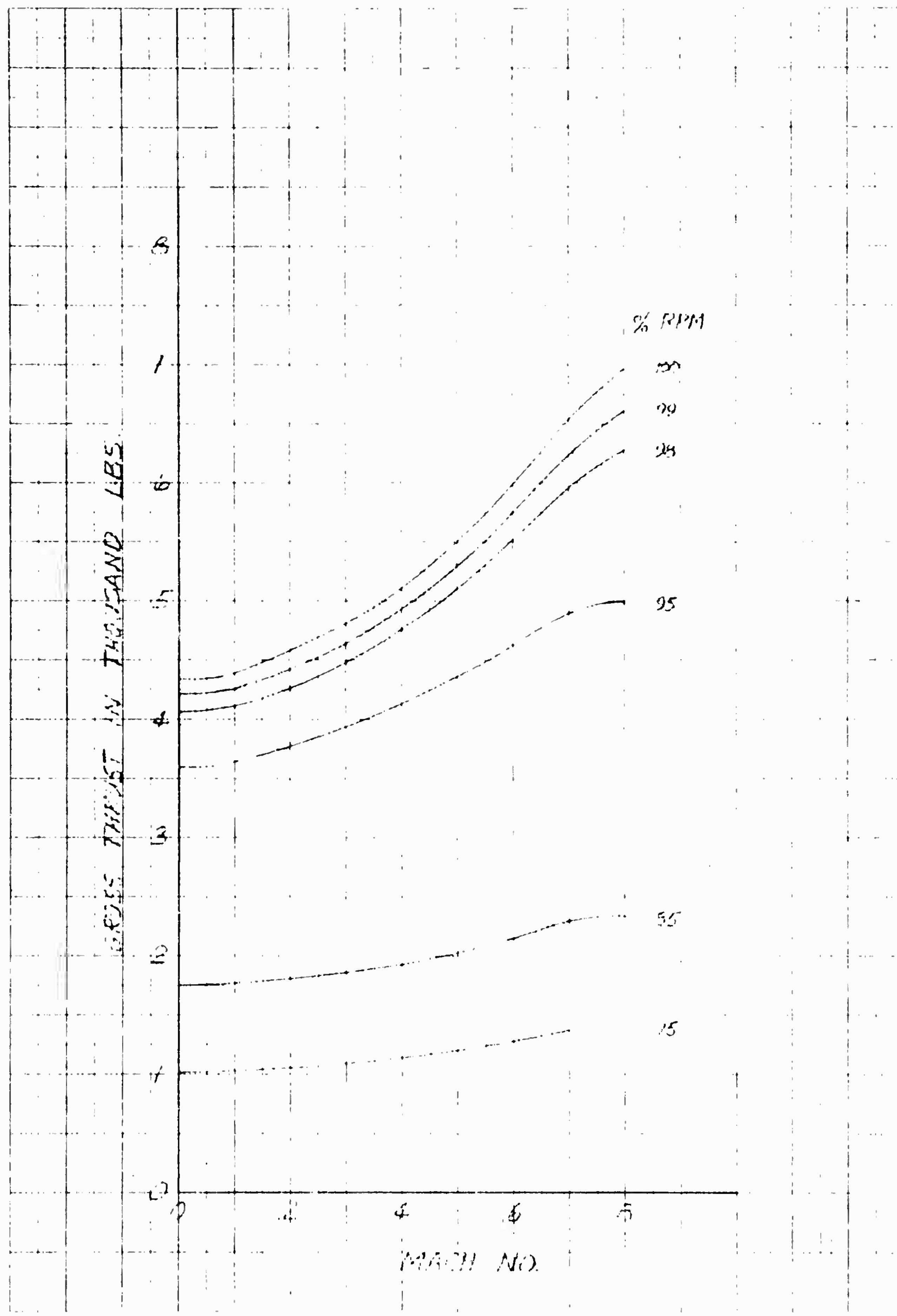

Figure 4.2 Gross Thrust vs Mach. No. and % RPM; Altitude = 5000 ft., 2 Engines, Standard Day

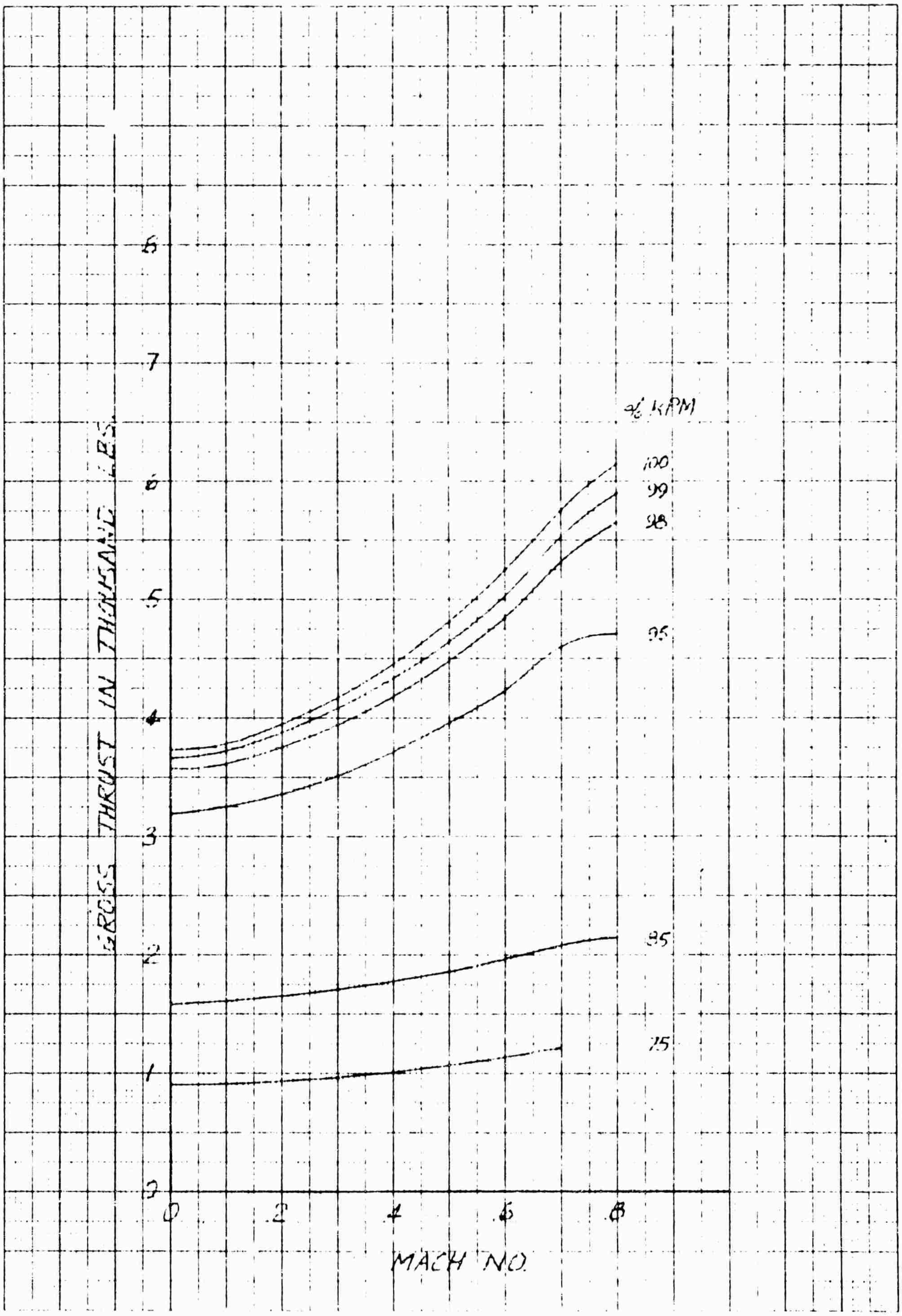

Figure 4.3 Gross Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 2Engines, Standard Day

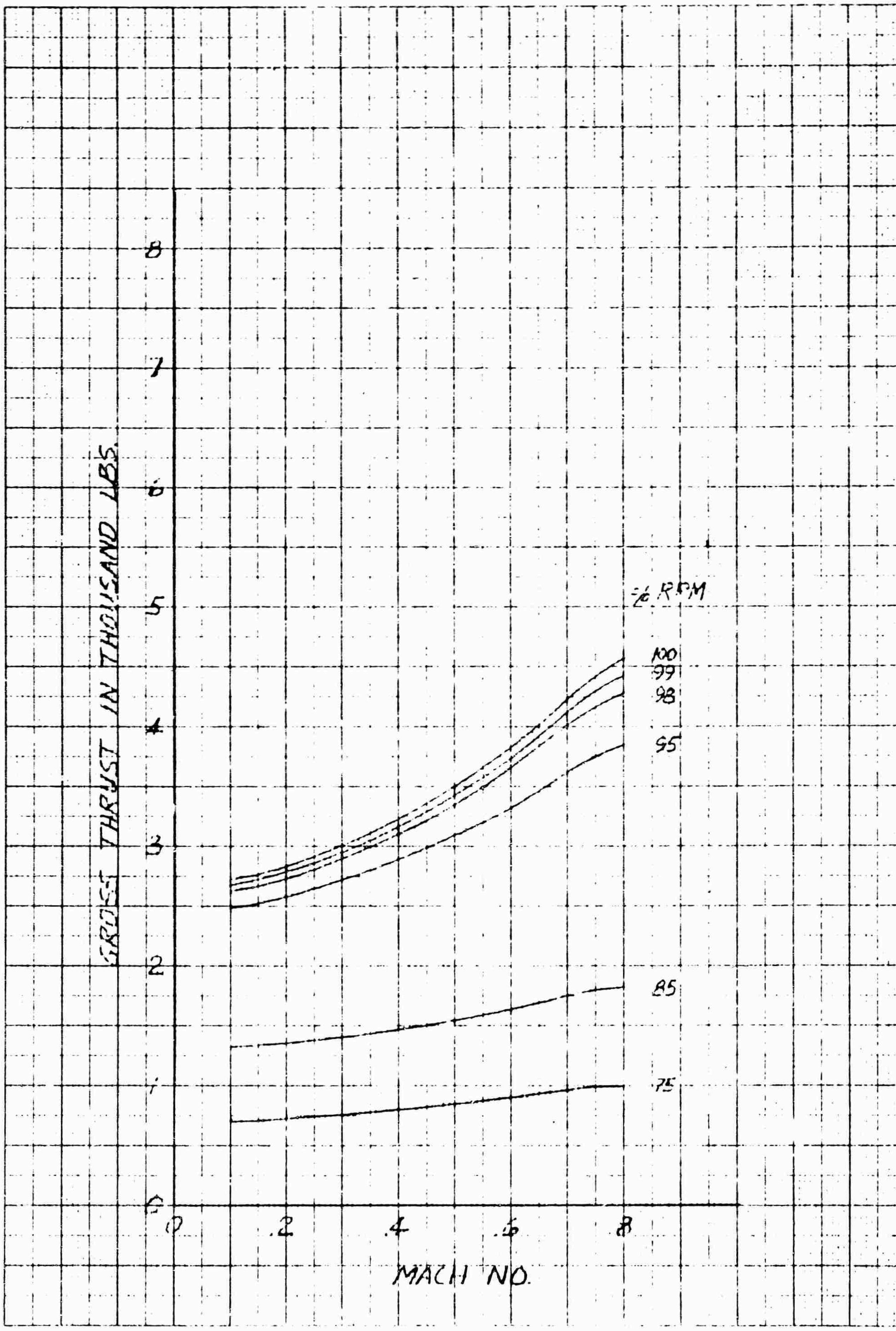

Figure 4.4 Gross Thrust vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Standard Day

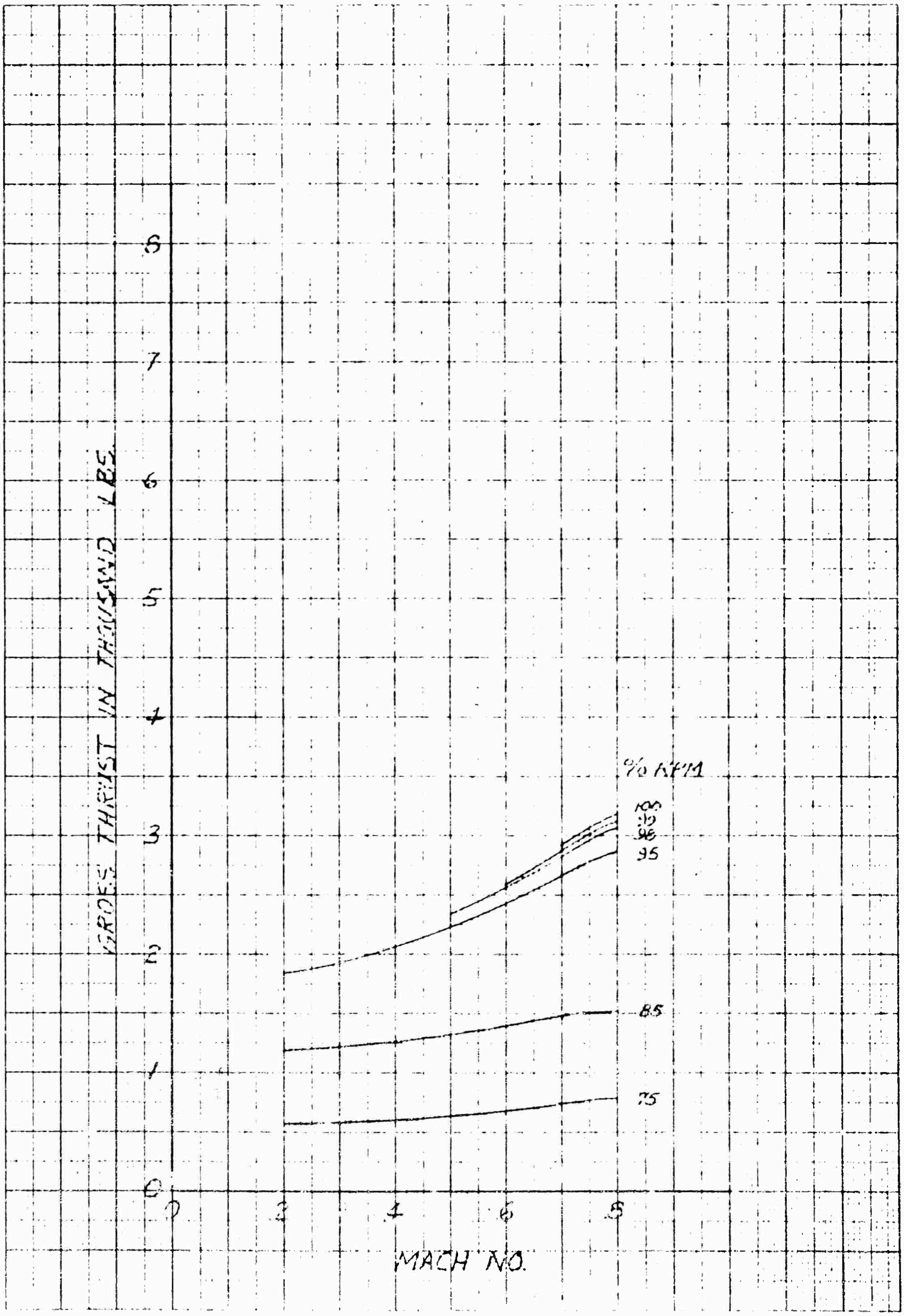

Figure 4.5 Gross Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Standard Day

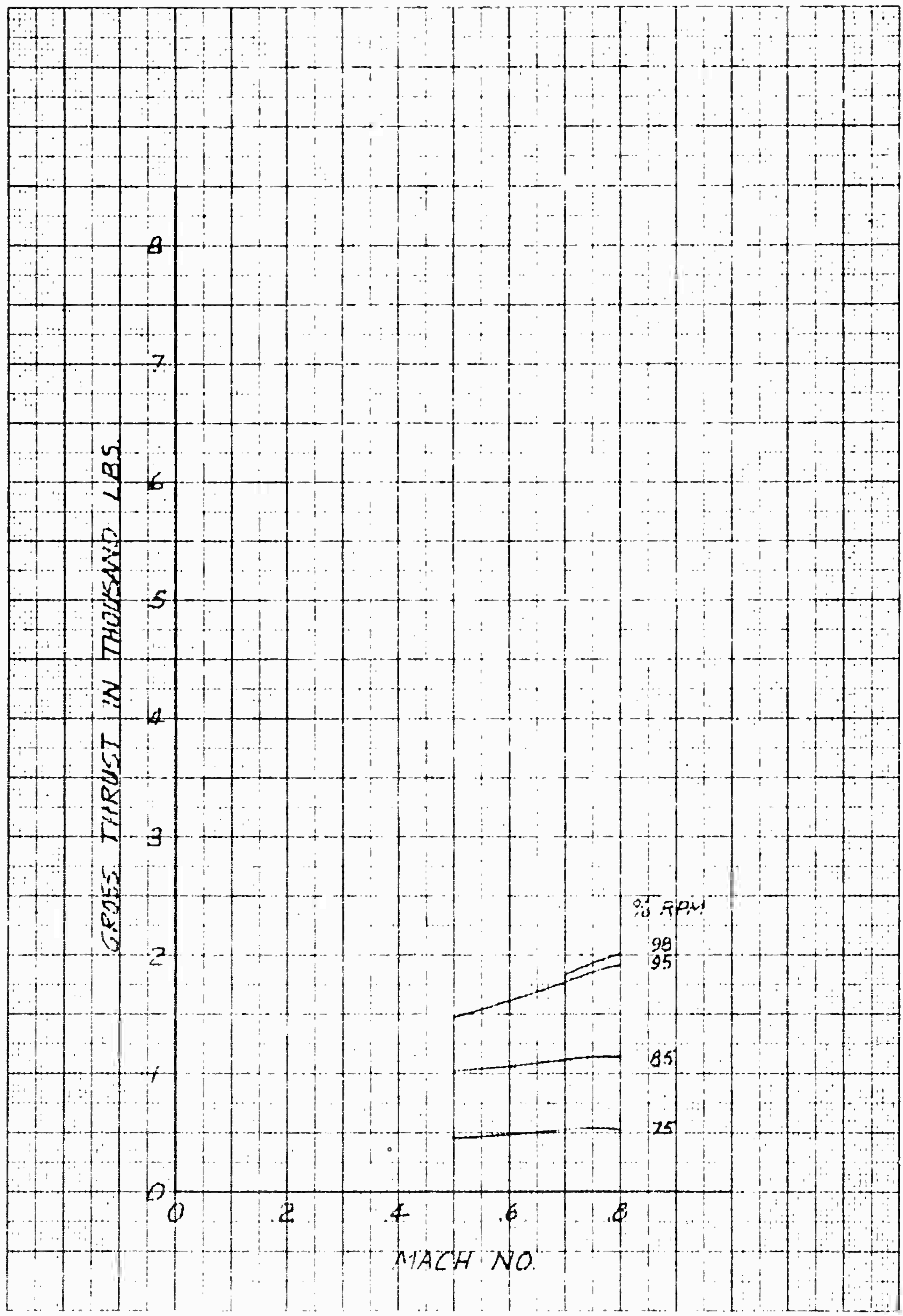

Figure 4.6 Gross Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Standard Day

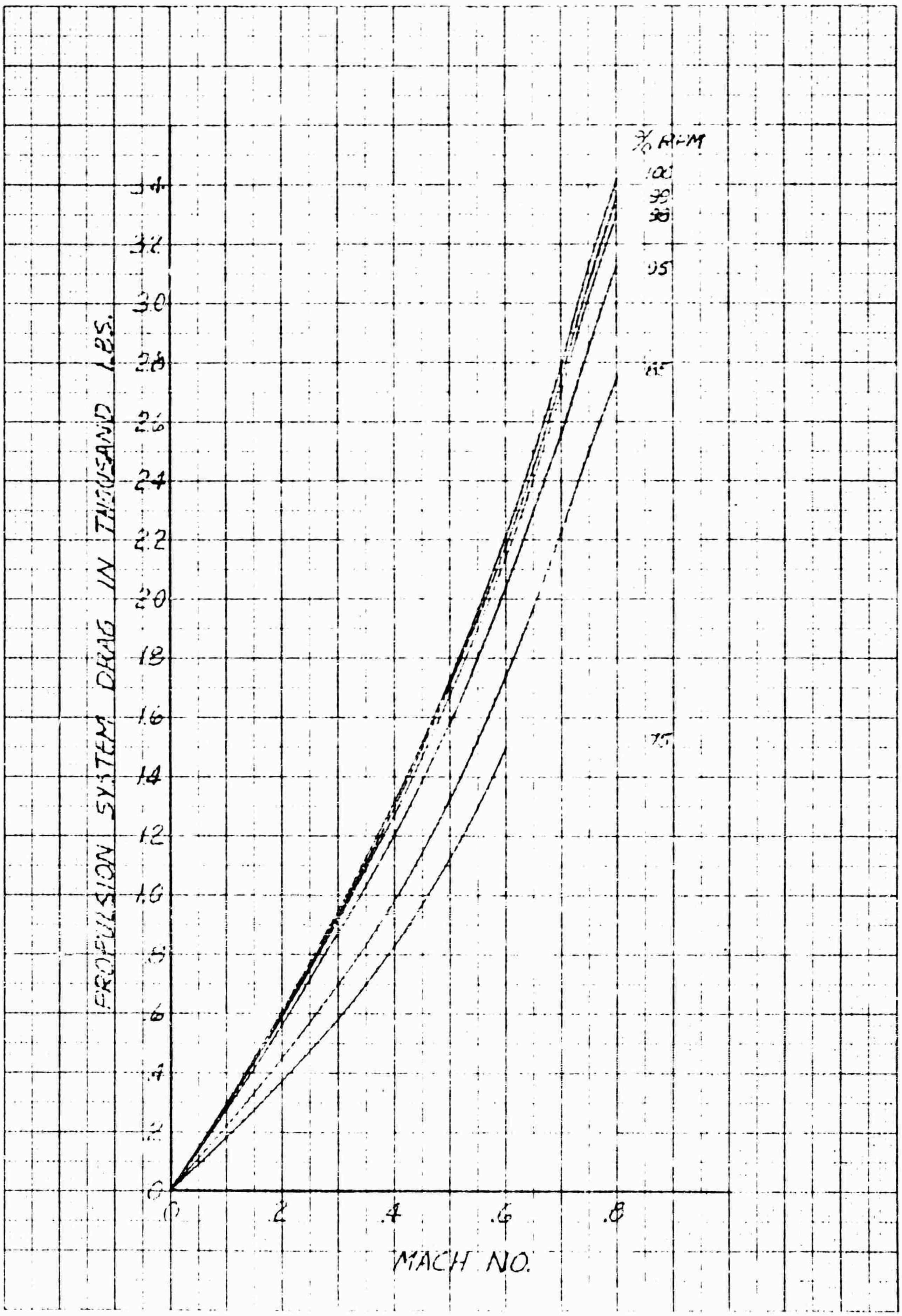

Figure 4.7 Propulsion System Drag vs Mach No. and % RPM;
Altitude = 0 ft., 2 Engines, Standard Day

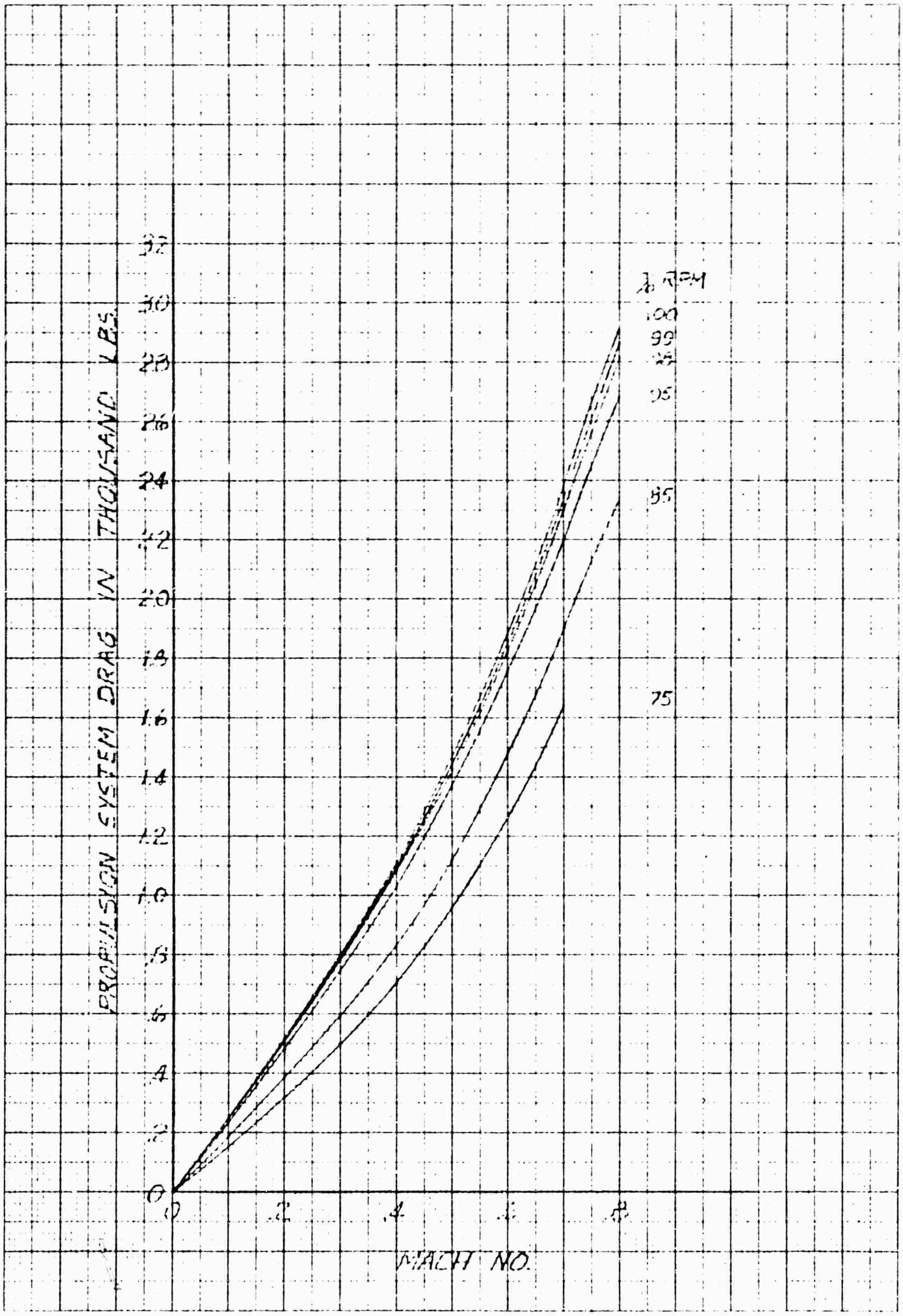

Figure 4.8 Propulsion System Drag vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Standard Day

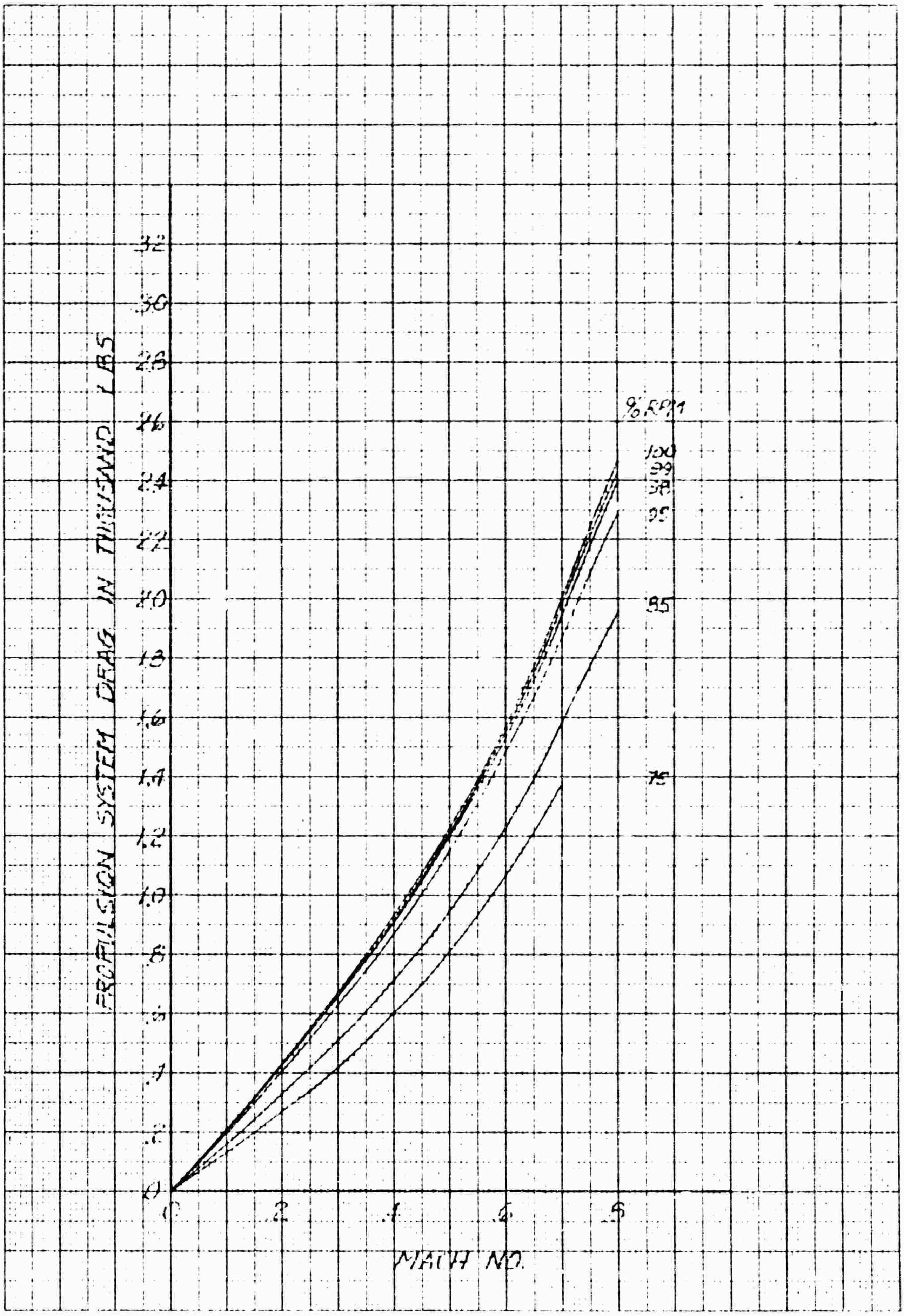

Figure 4.9 Propulsion System Drag vs Mach No. and % RPM;
Altitude = 10,000 ft., 2 Engines, Standard Day

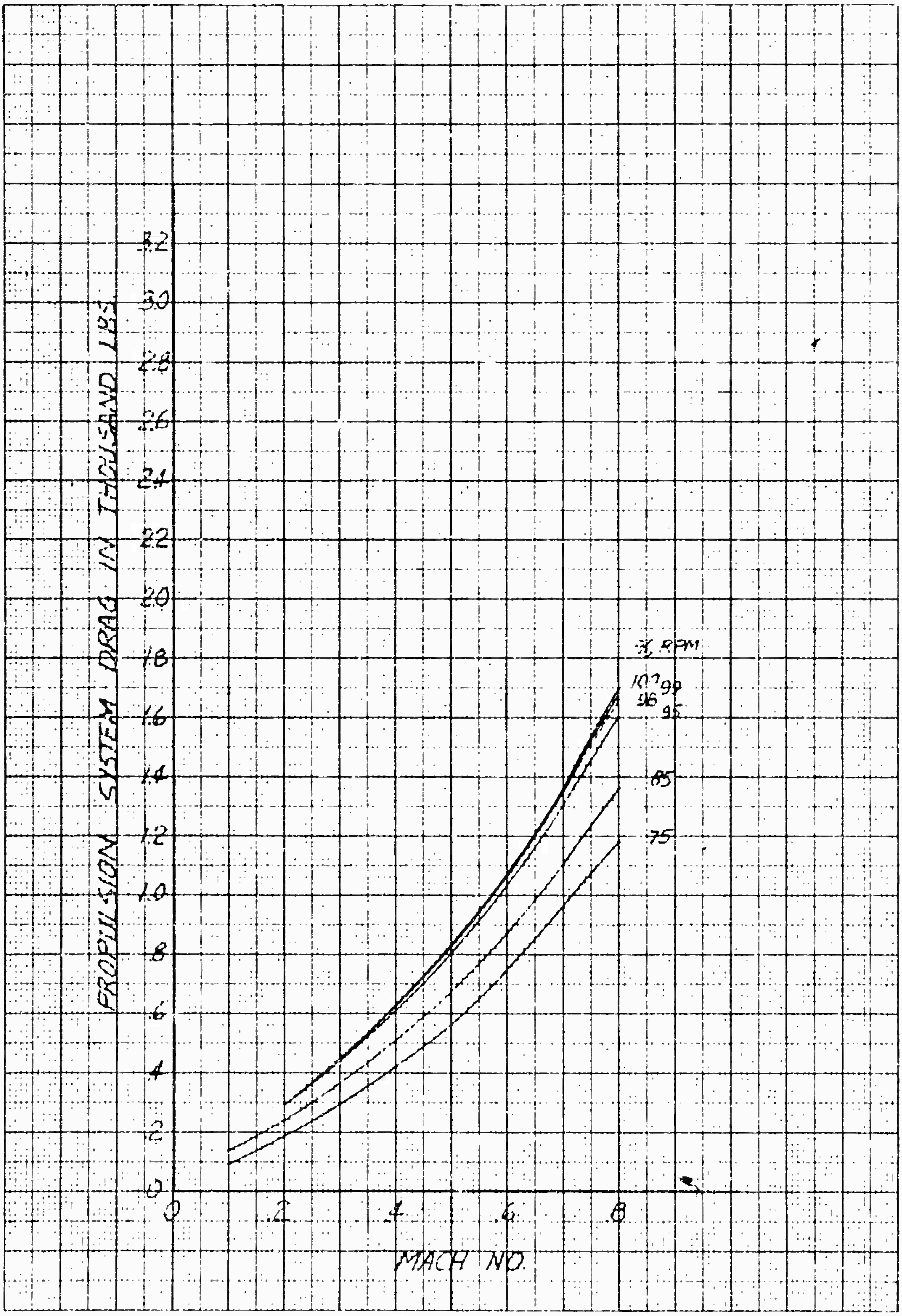

Figure 4.10 Propulsion System Drag vs Mach No. and % RPM;
Altitude = 20,000 ft., 2 Engines, Standard Day

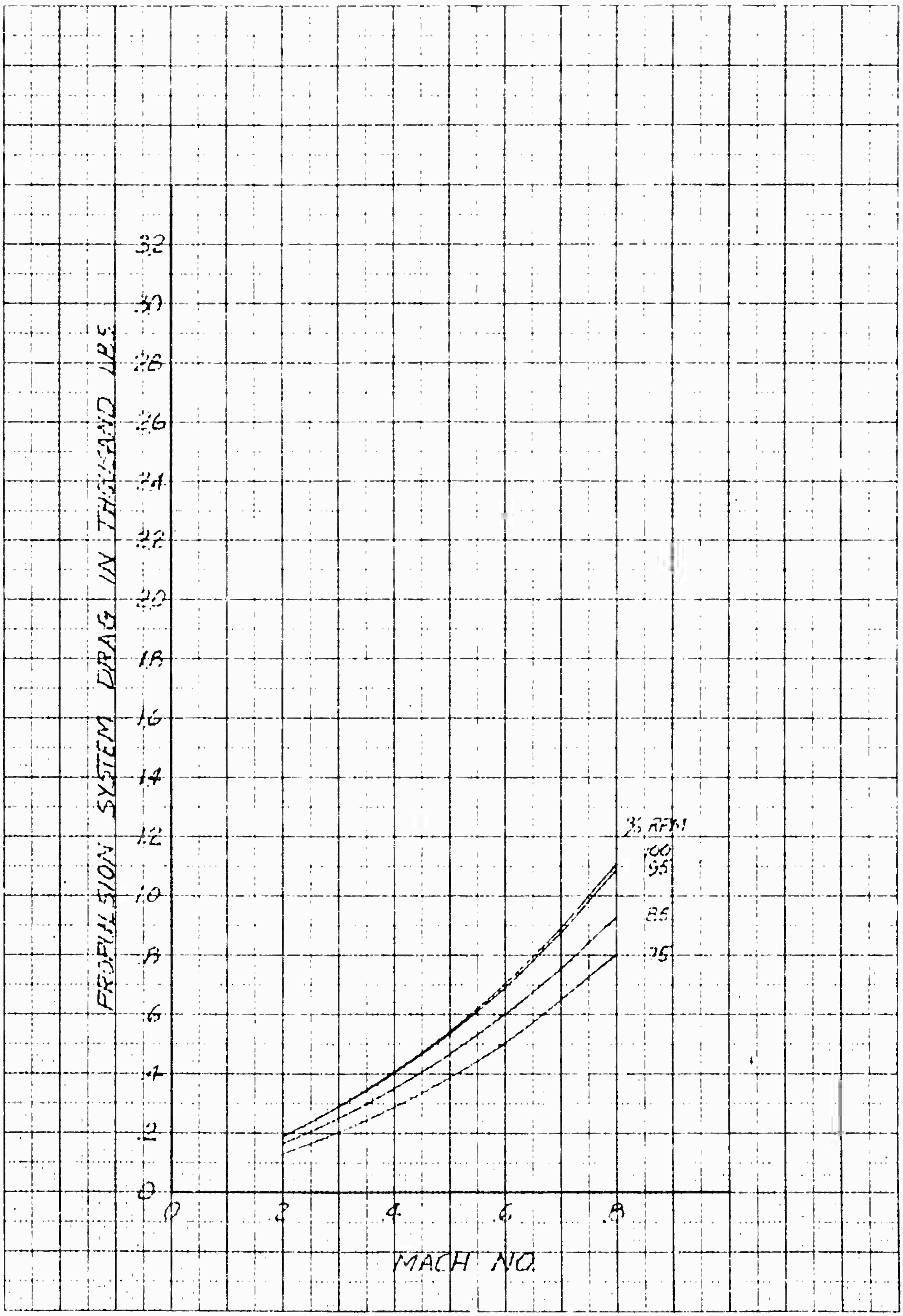

Figure 4.11 Propulsion System Drag vs Mach No. and % RPM; Altitude = 30,000 ft. 2 Engines, Standard Day

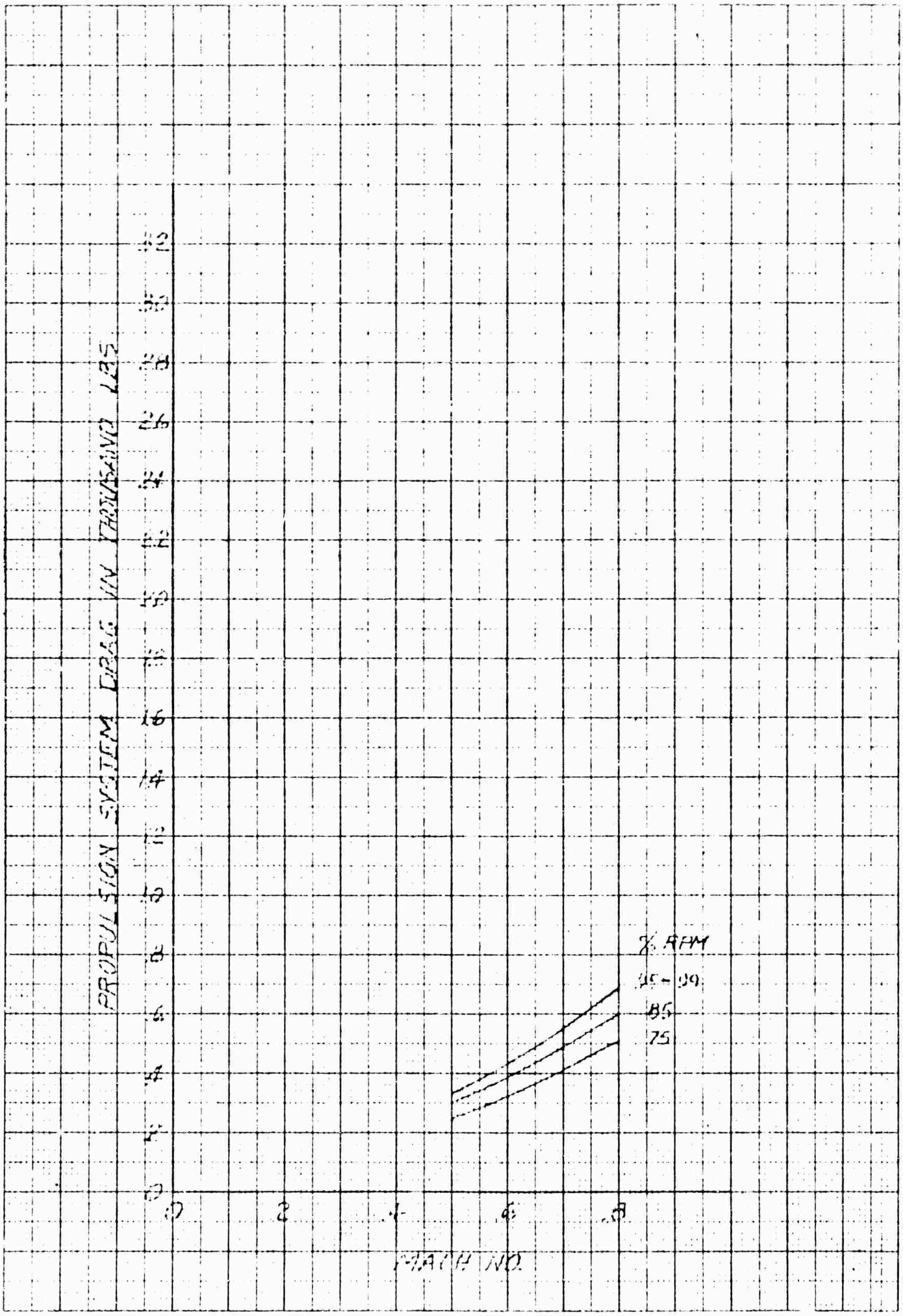

Figure 4.12 Propulsion System Drag vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Standard Day

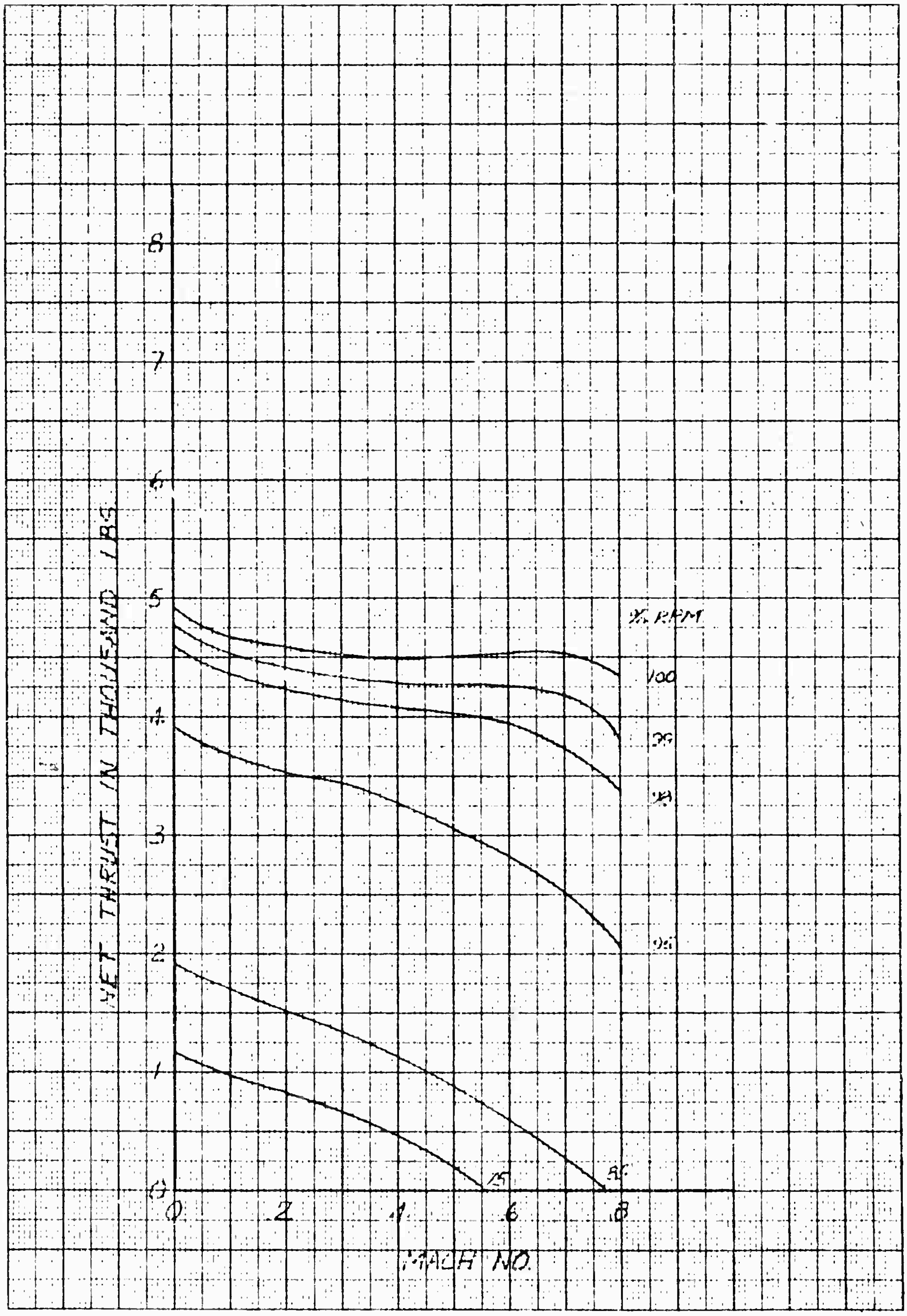

Figure 4.13 Net Thrust vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Standard Day

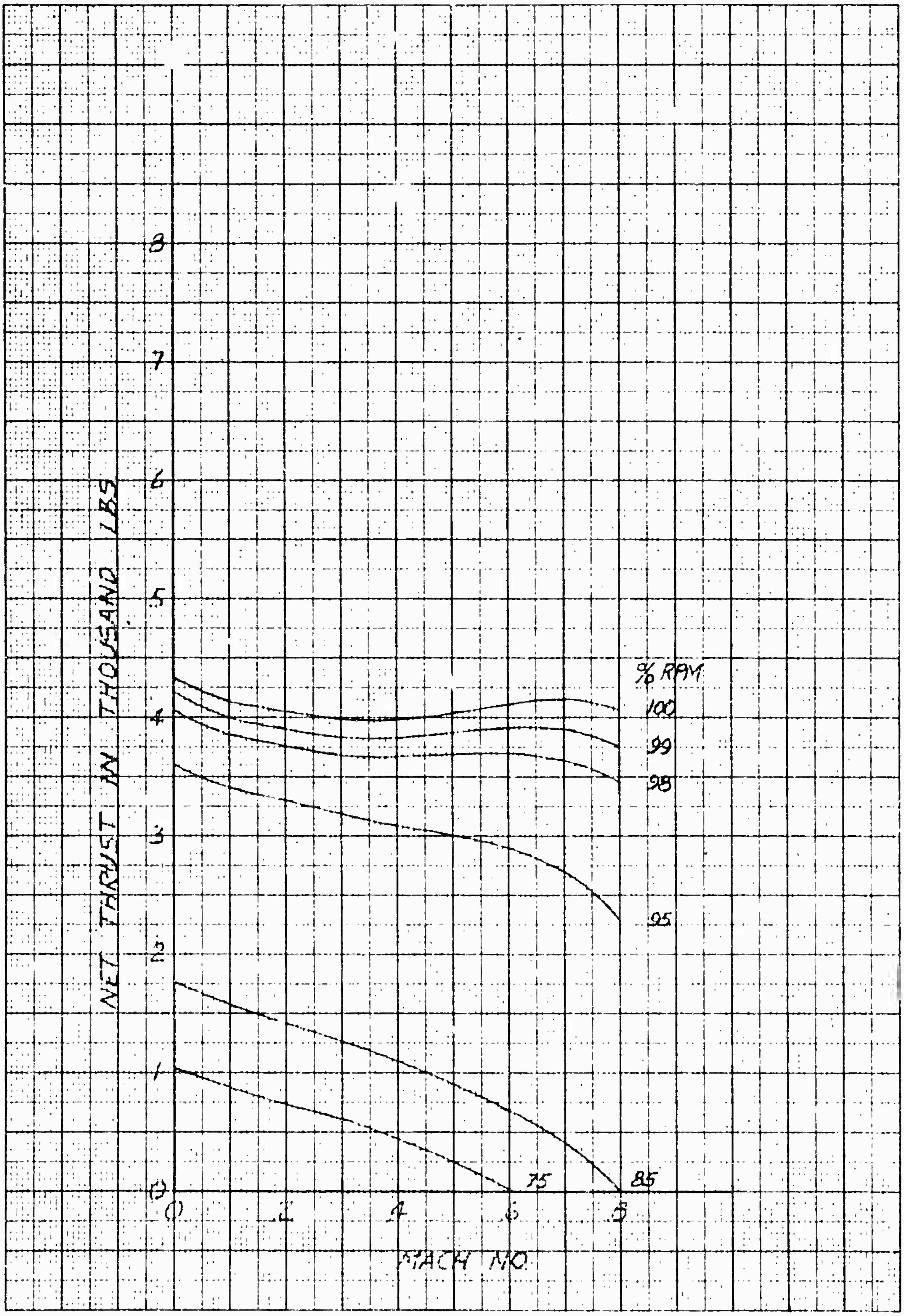

Figure 4.14 Net Thrust vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Standard Day

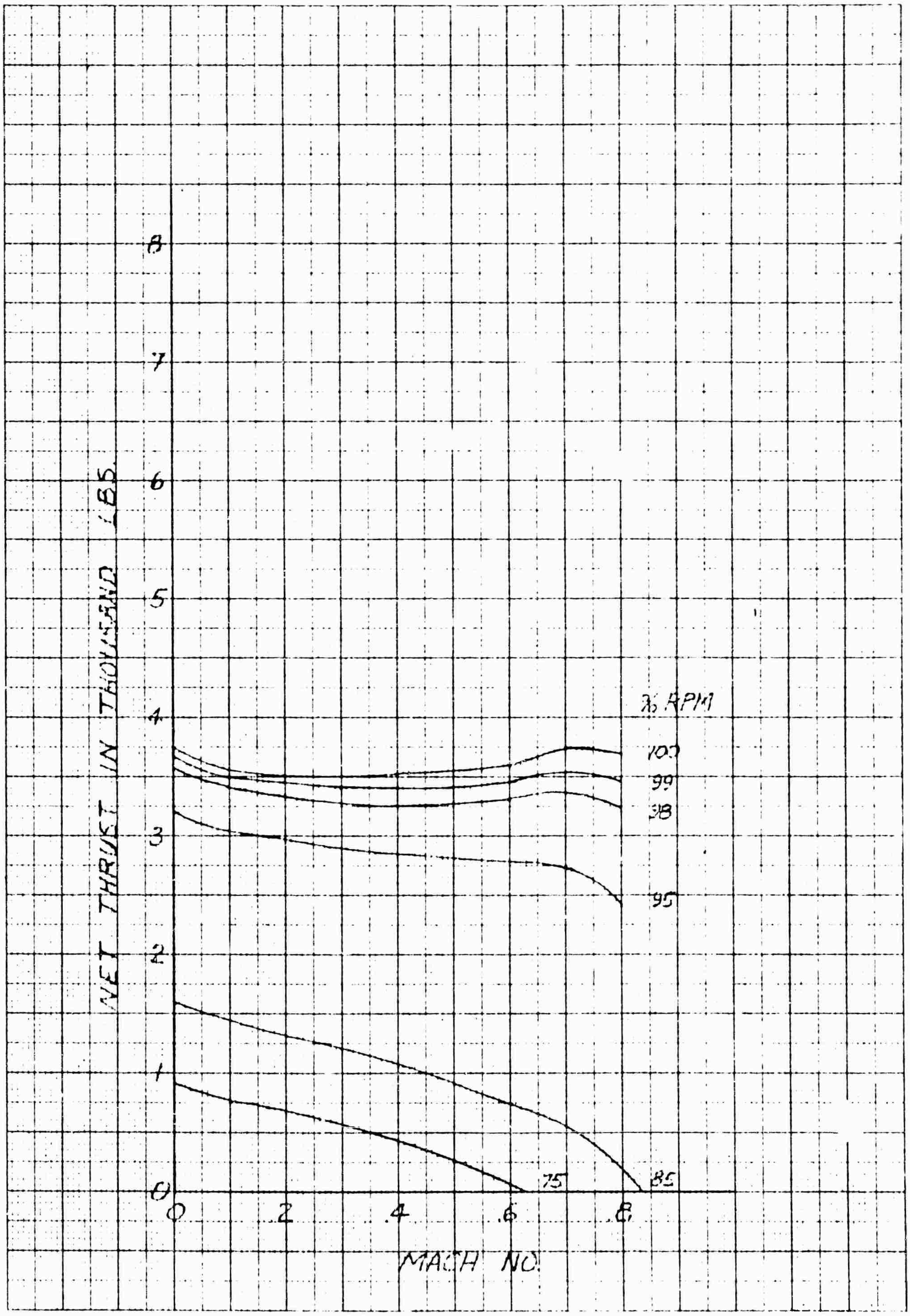

Figure 4.15 Net Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Standard Day

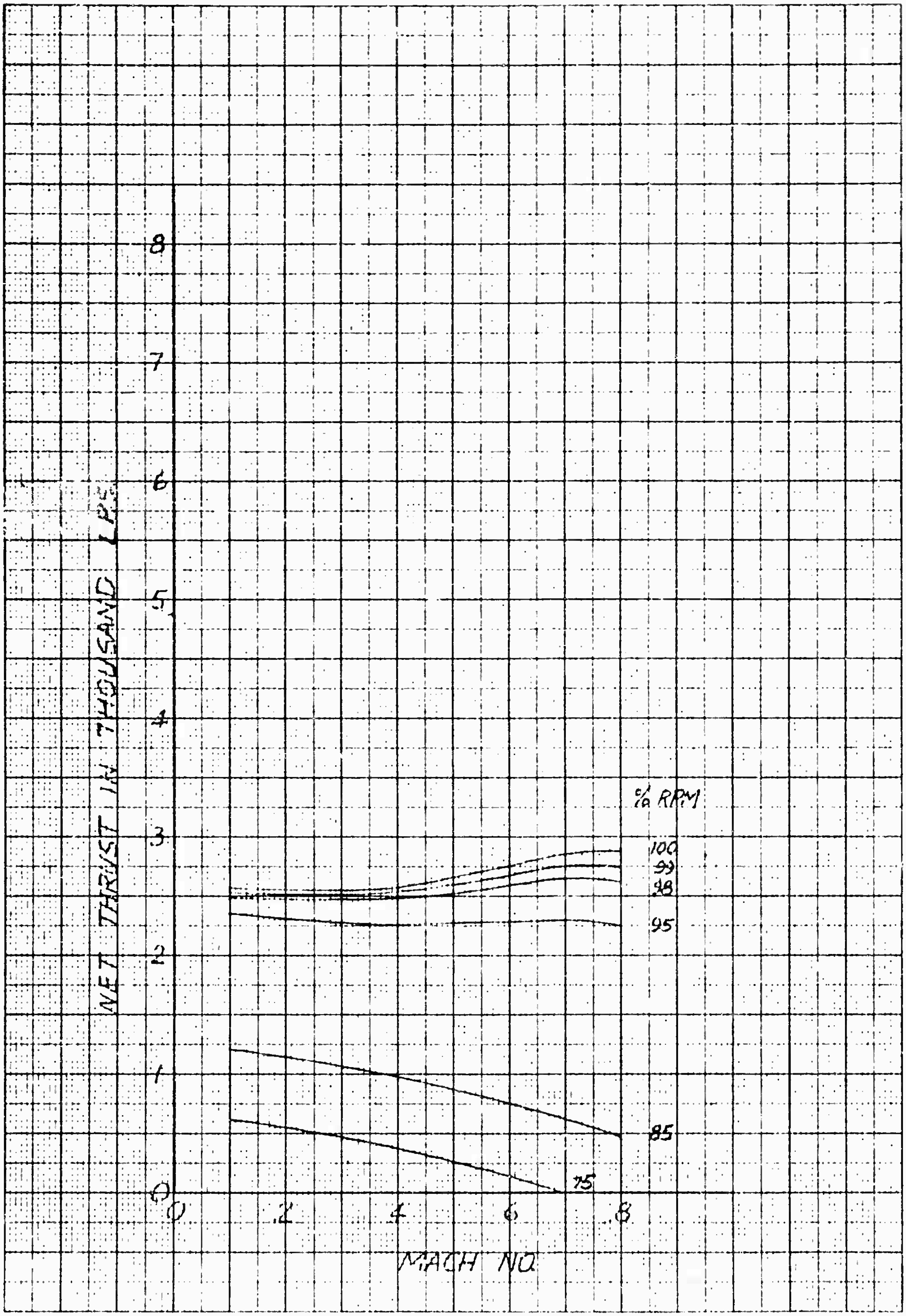

Figure 4.16 Net Thrust vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Standard Day

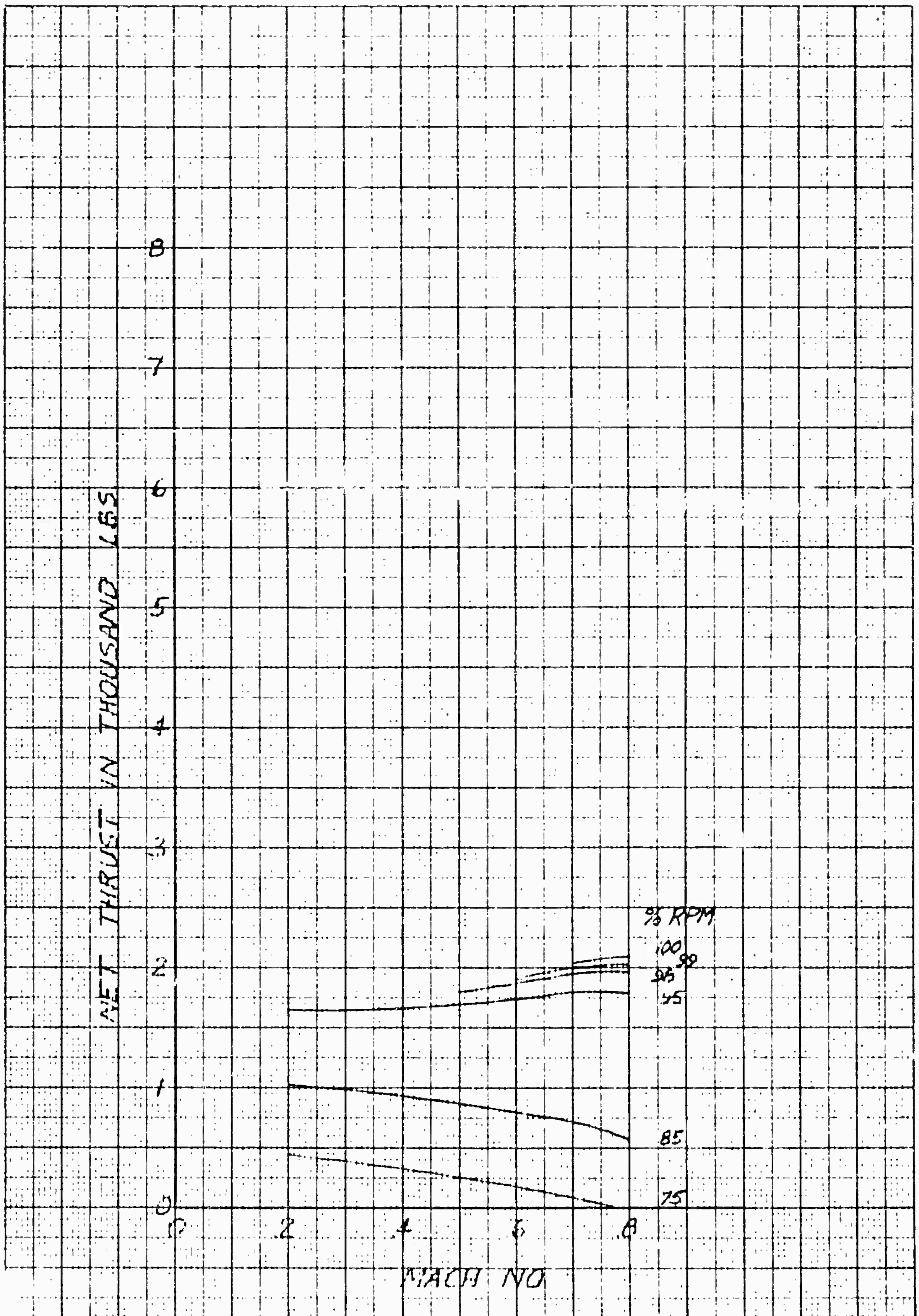

Figure 4.17 Net Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Standard Day

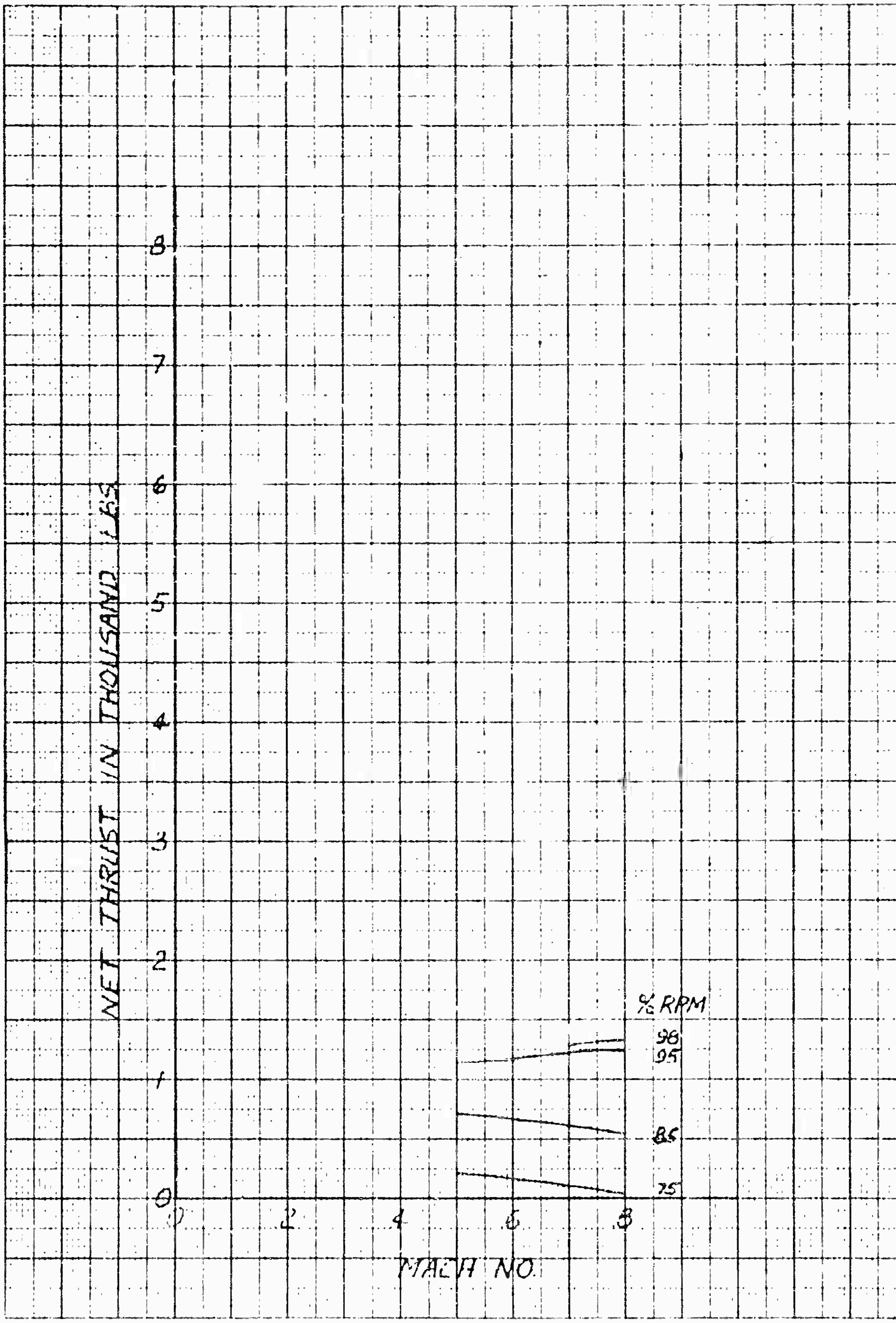

Figure 4.18 Net Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Standard Day

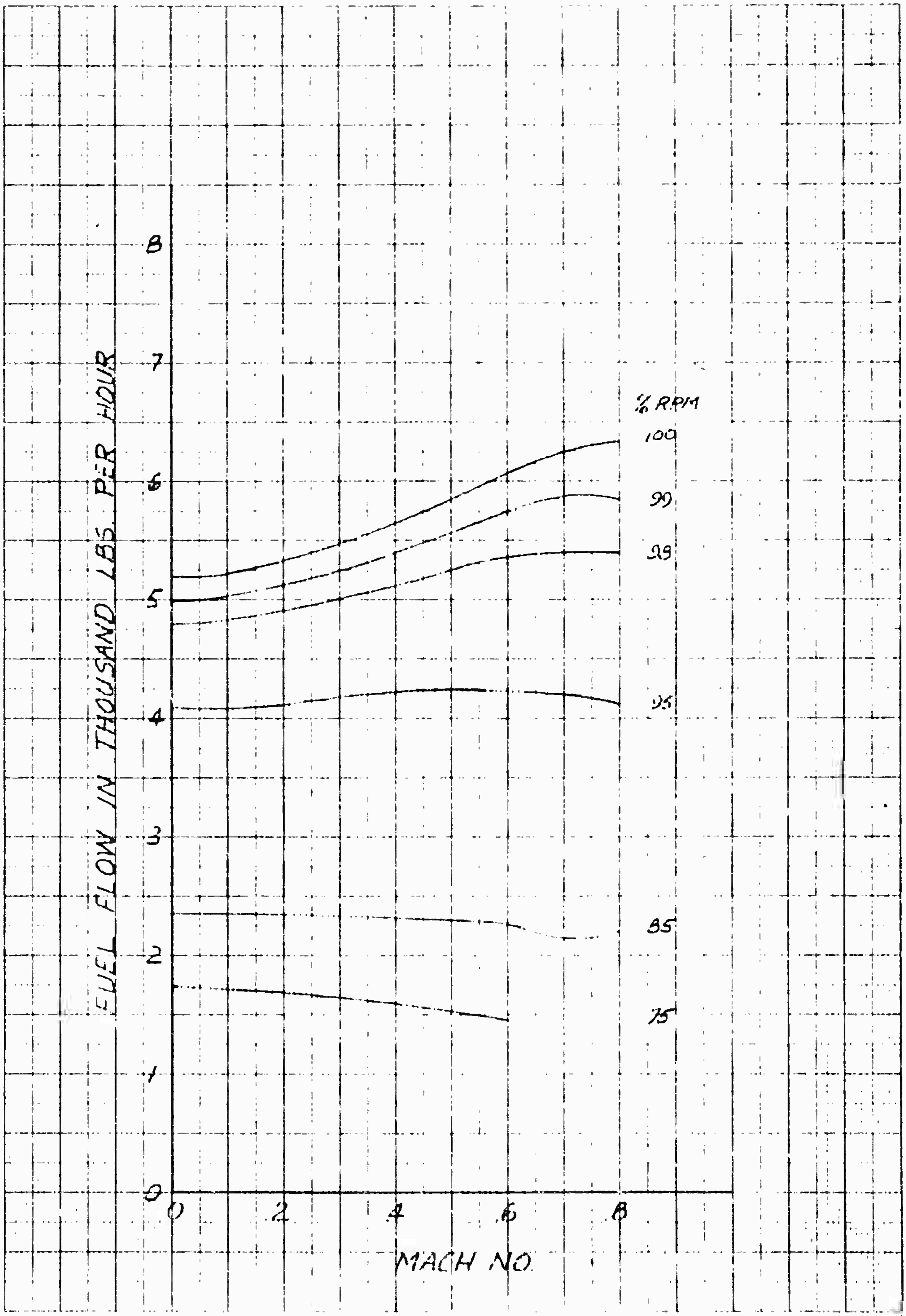

Figure 4.19 Fuel Flow vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Standard Day

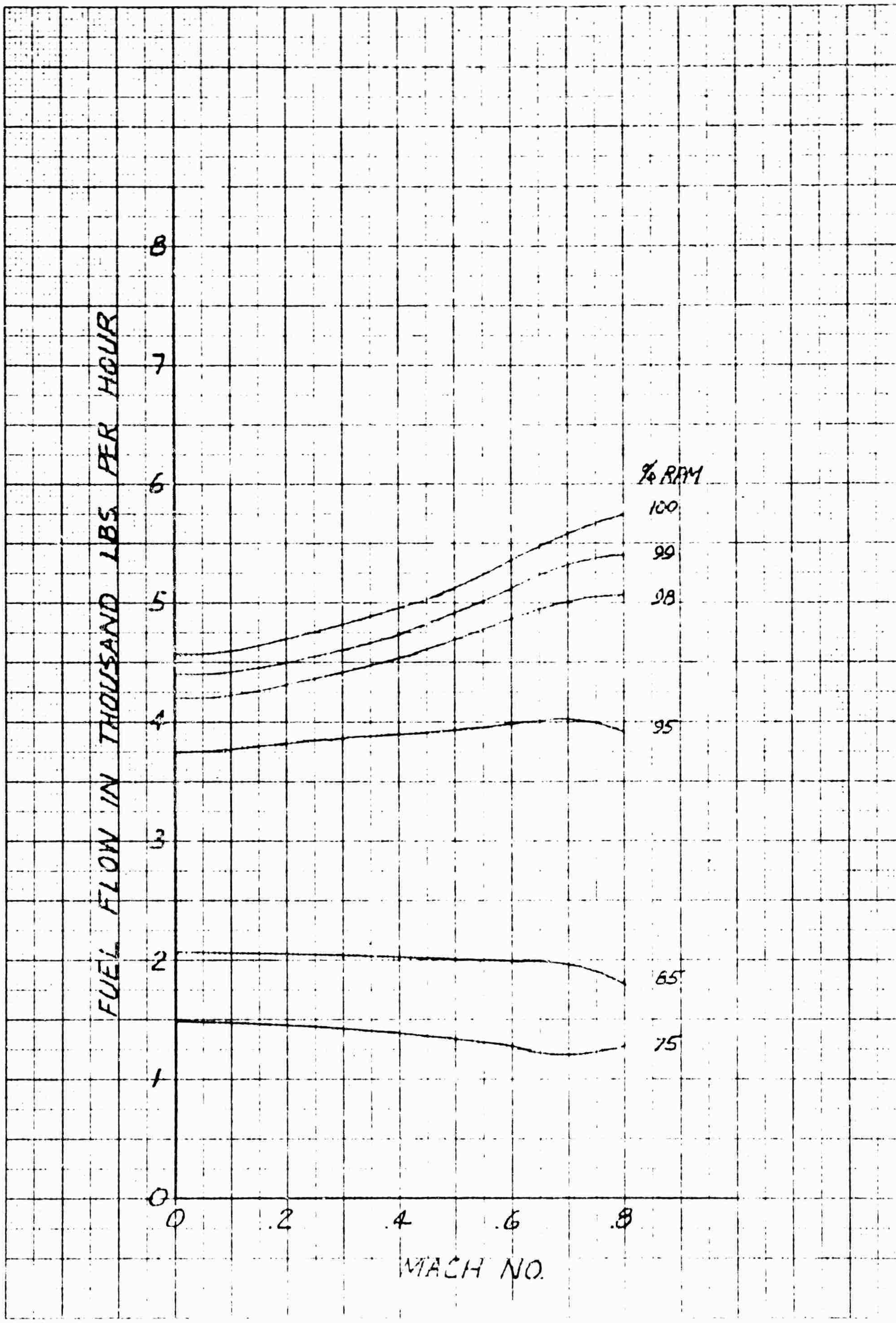

Figure 4.20 Fuel Flow vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Standard Day

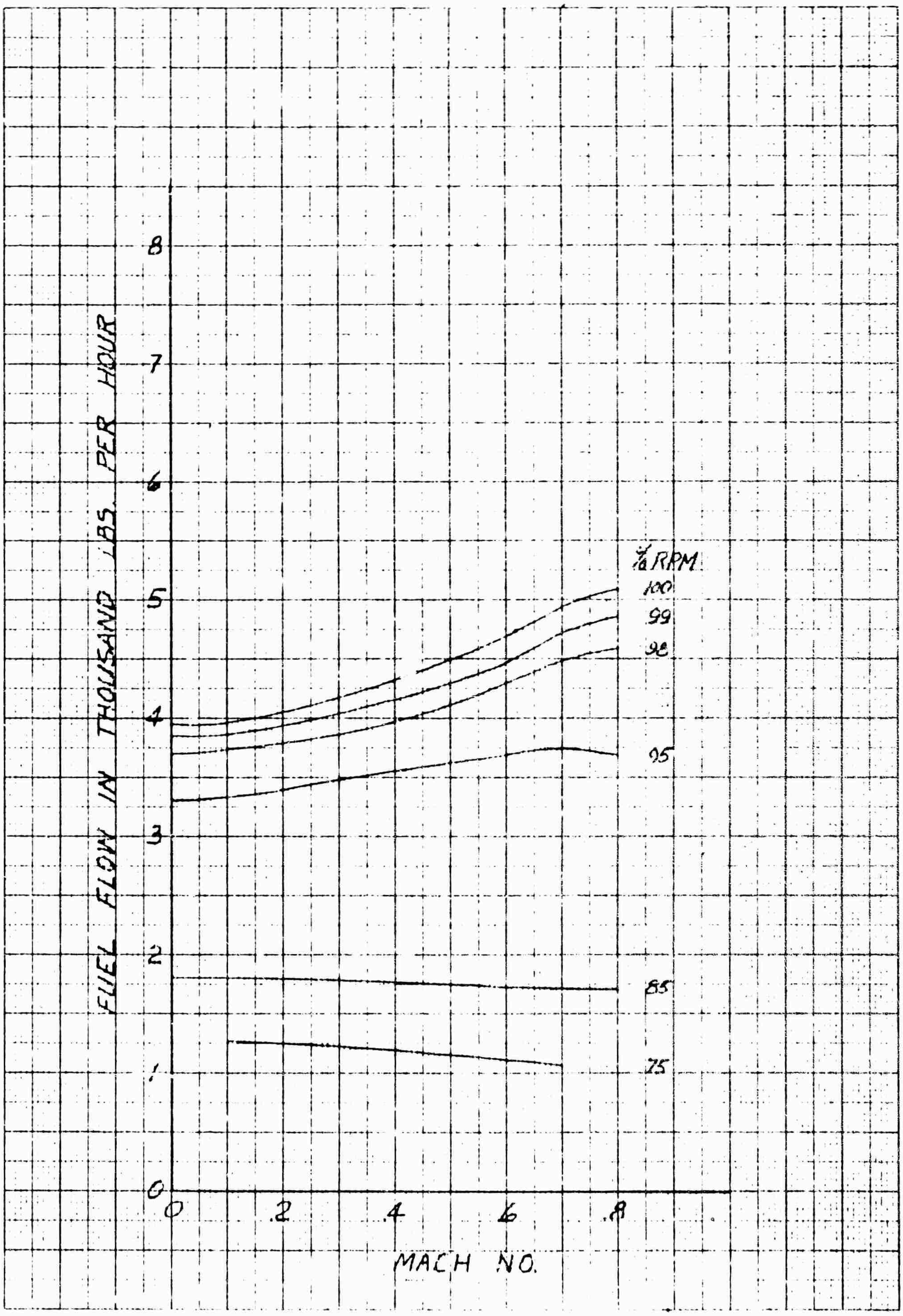

Figure 4.21 Fuel Flow vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Standard Day

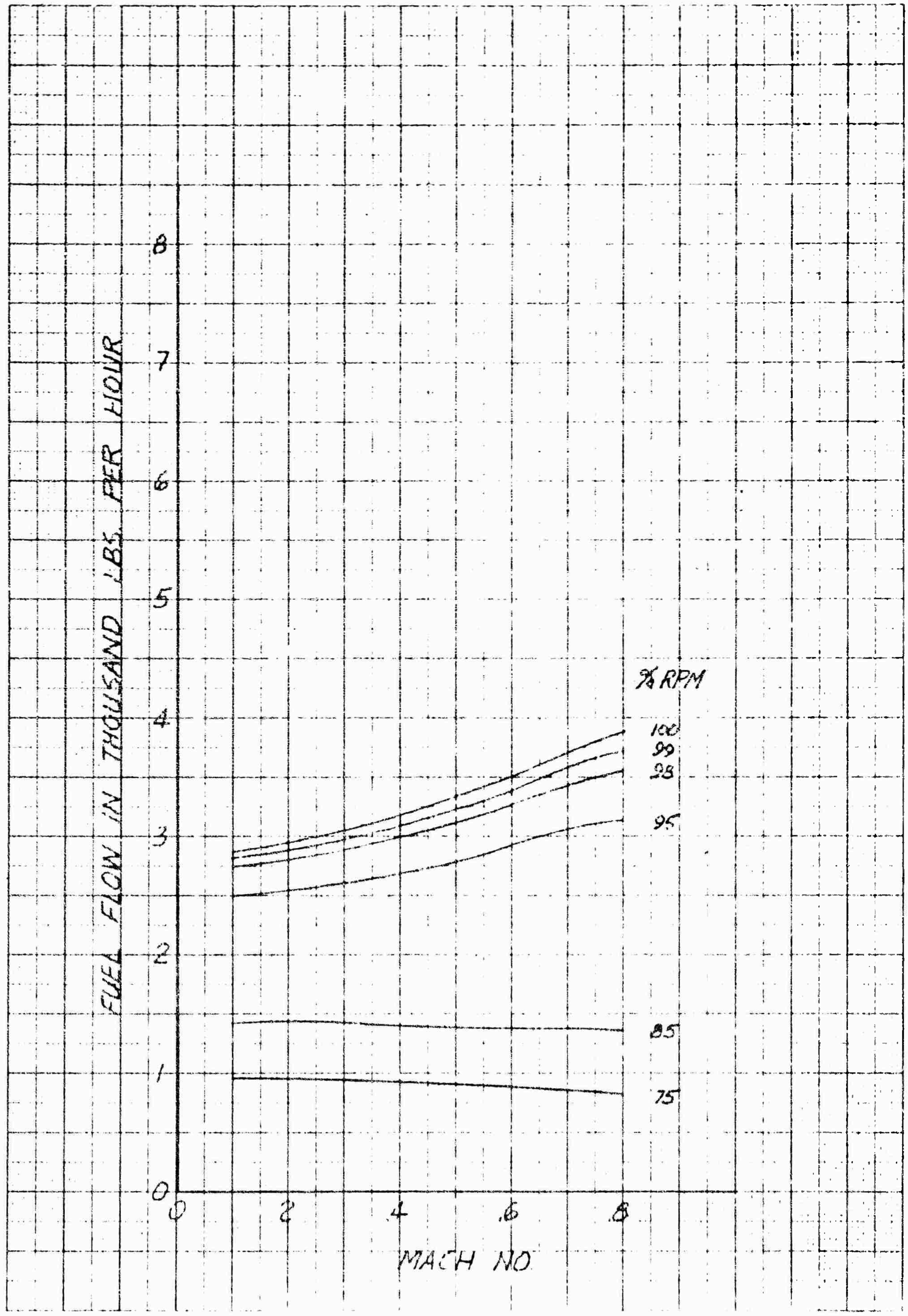

Figure 4.22 Fuel Flow vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Standard Day

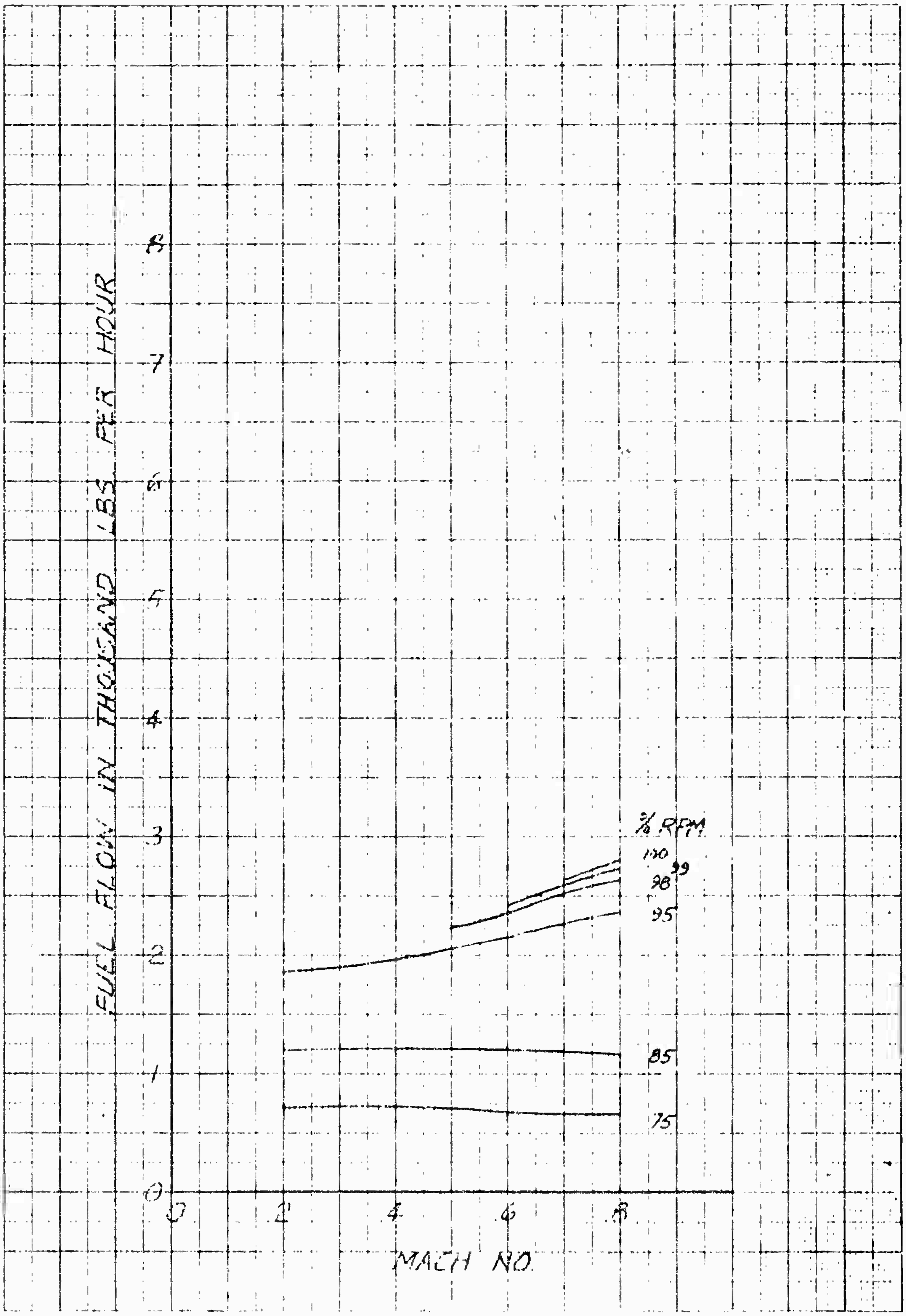

Figure 4.23 Fuel Flow vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Standard Day

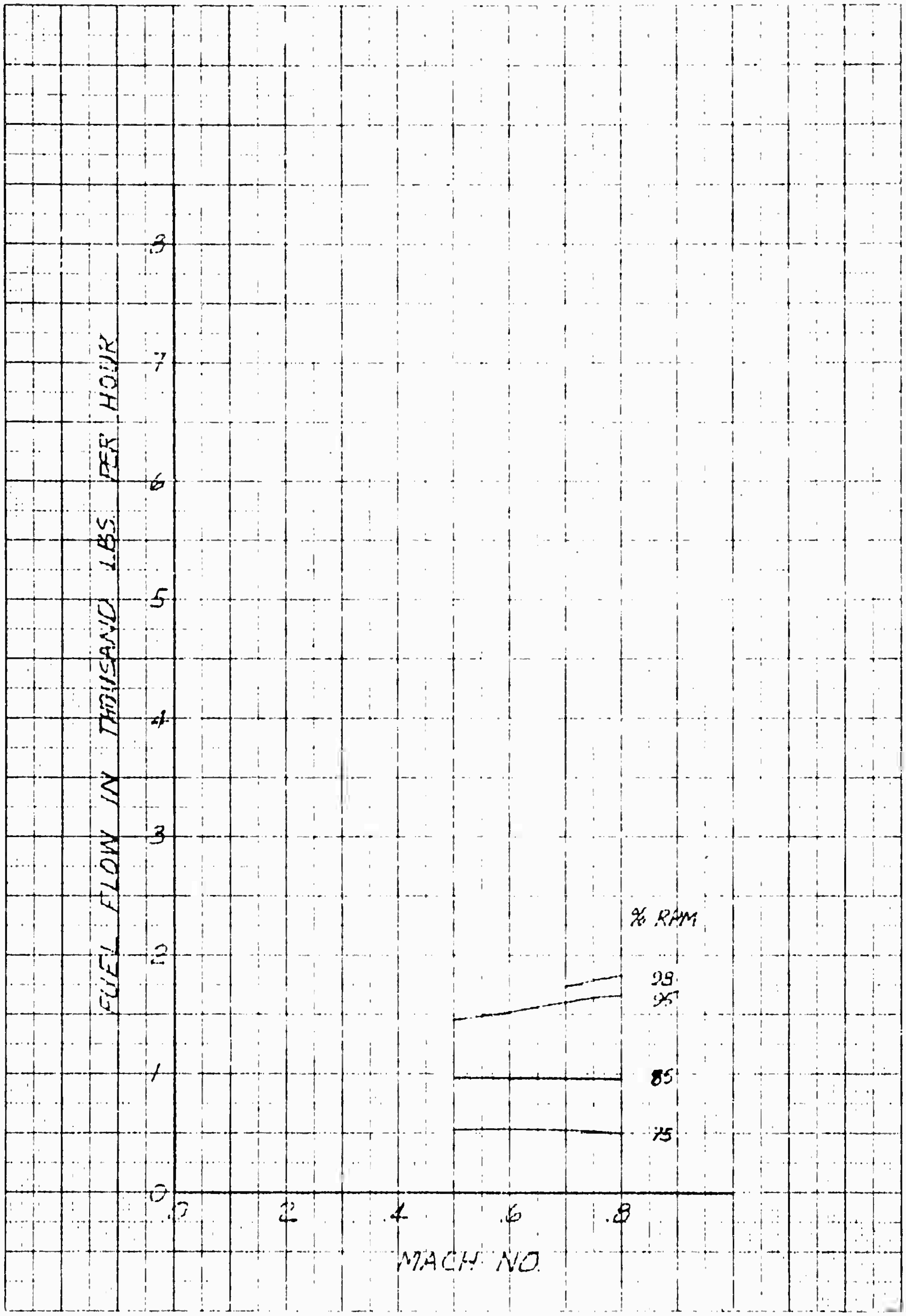

Figure 4.24 Fuel Flow vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Standard Day

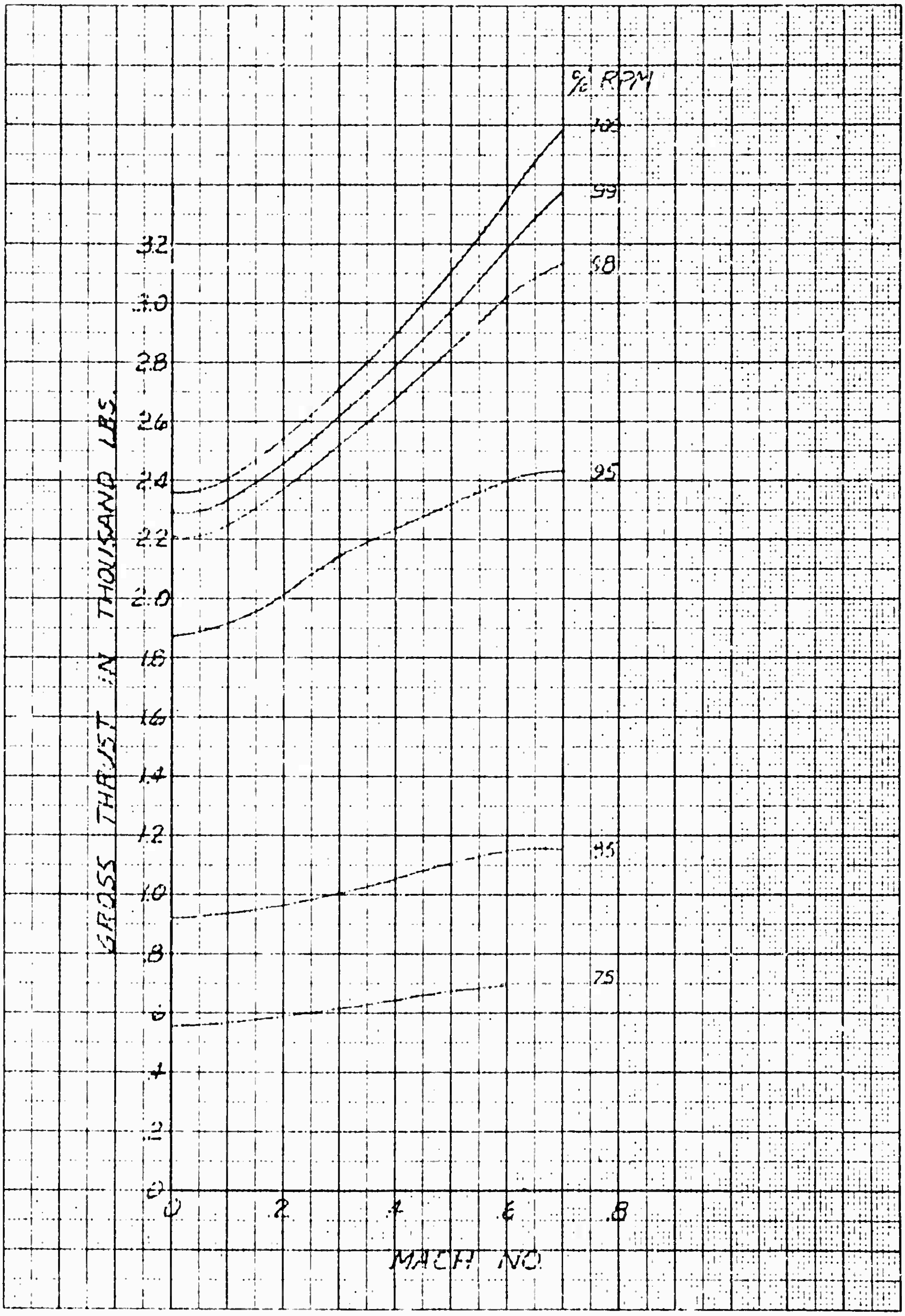

Figure 4.25 Gross Thrust vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Standard Day

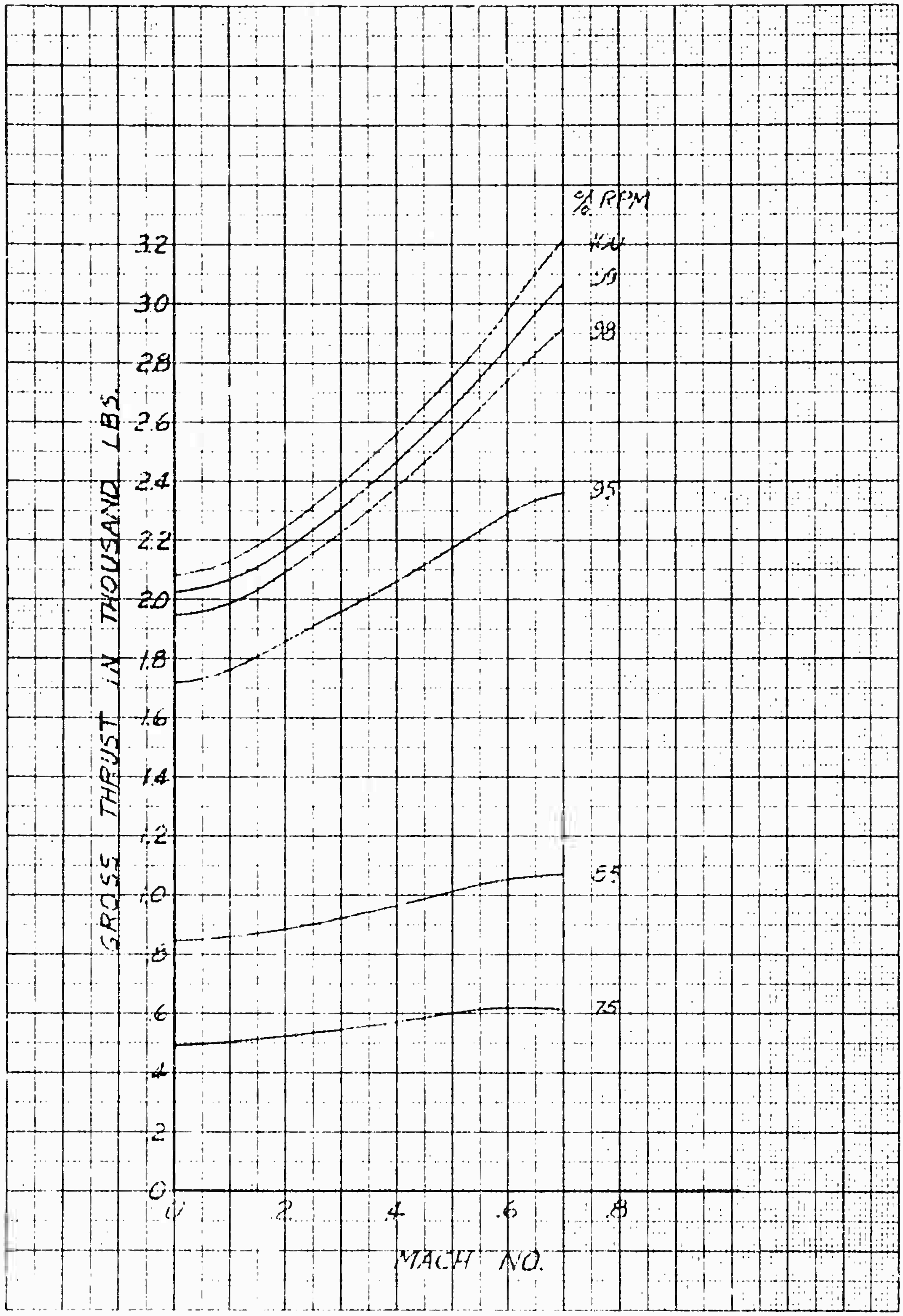

Figure 4.26 Gross Thrust vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Standard Day

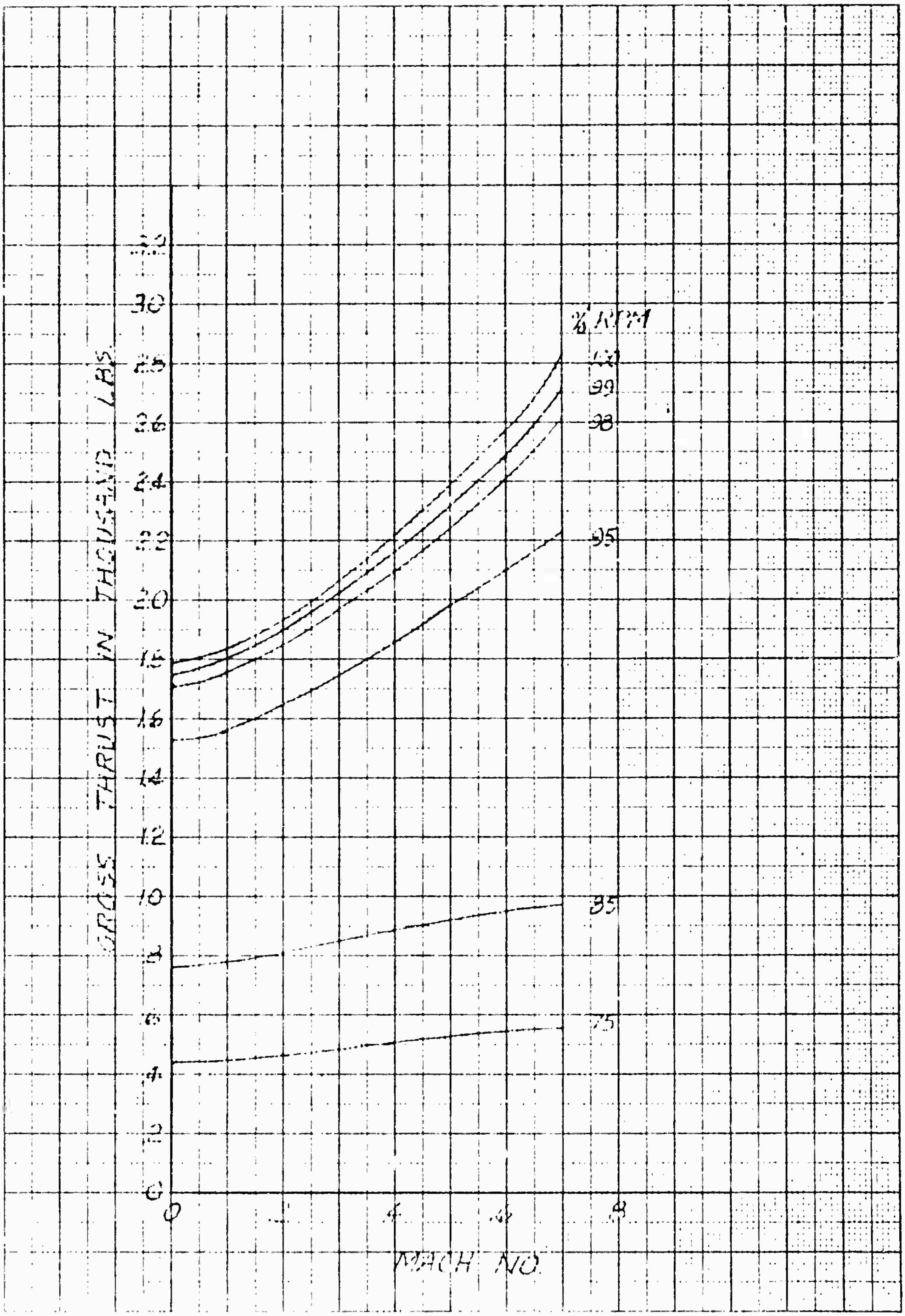

Figure 4.27 Gross Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Standard Day

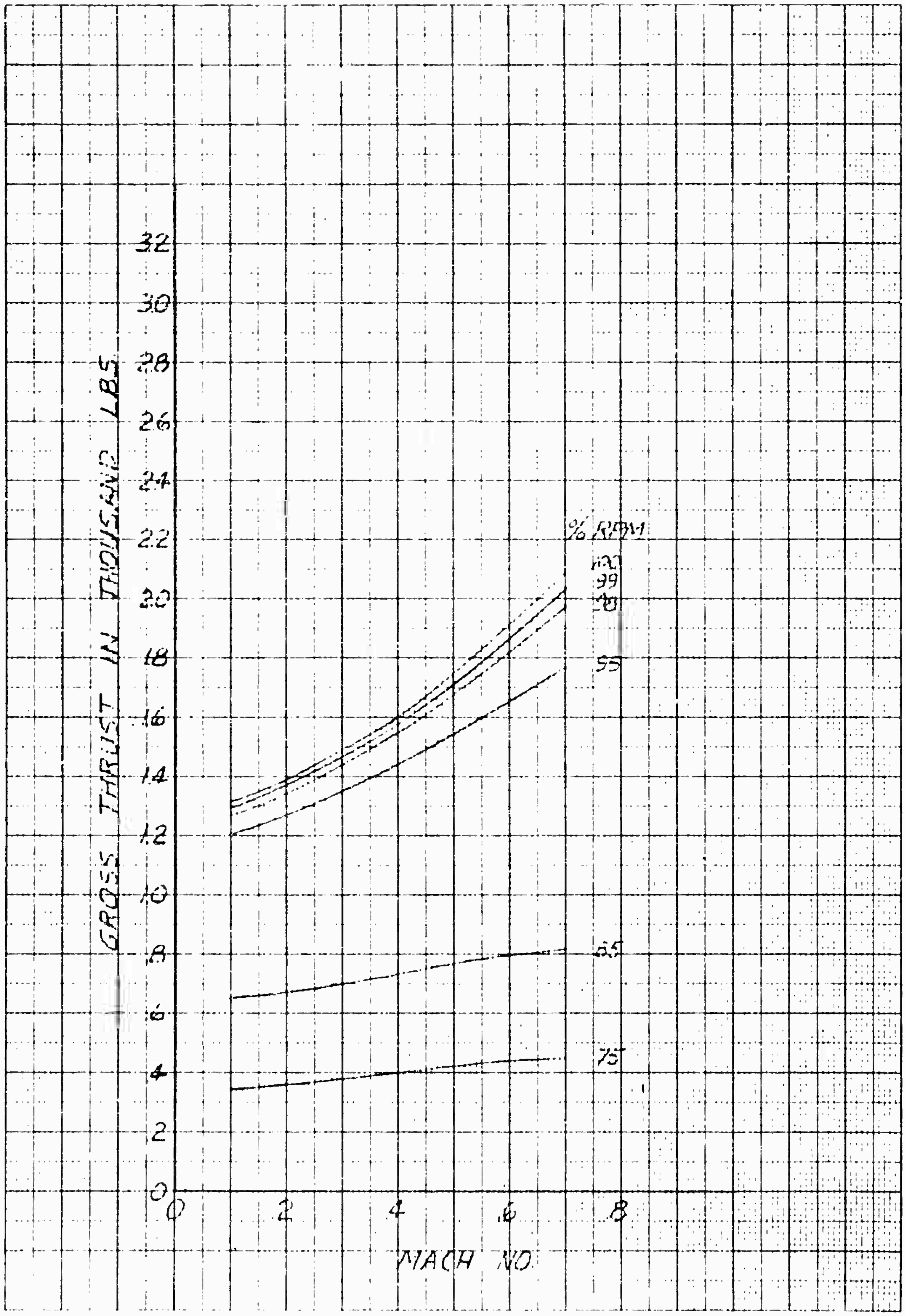

Figure 4.28 Gross Thrust vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Standard Day

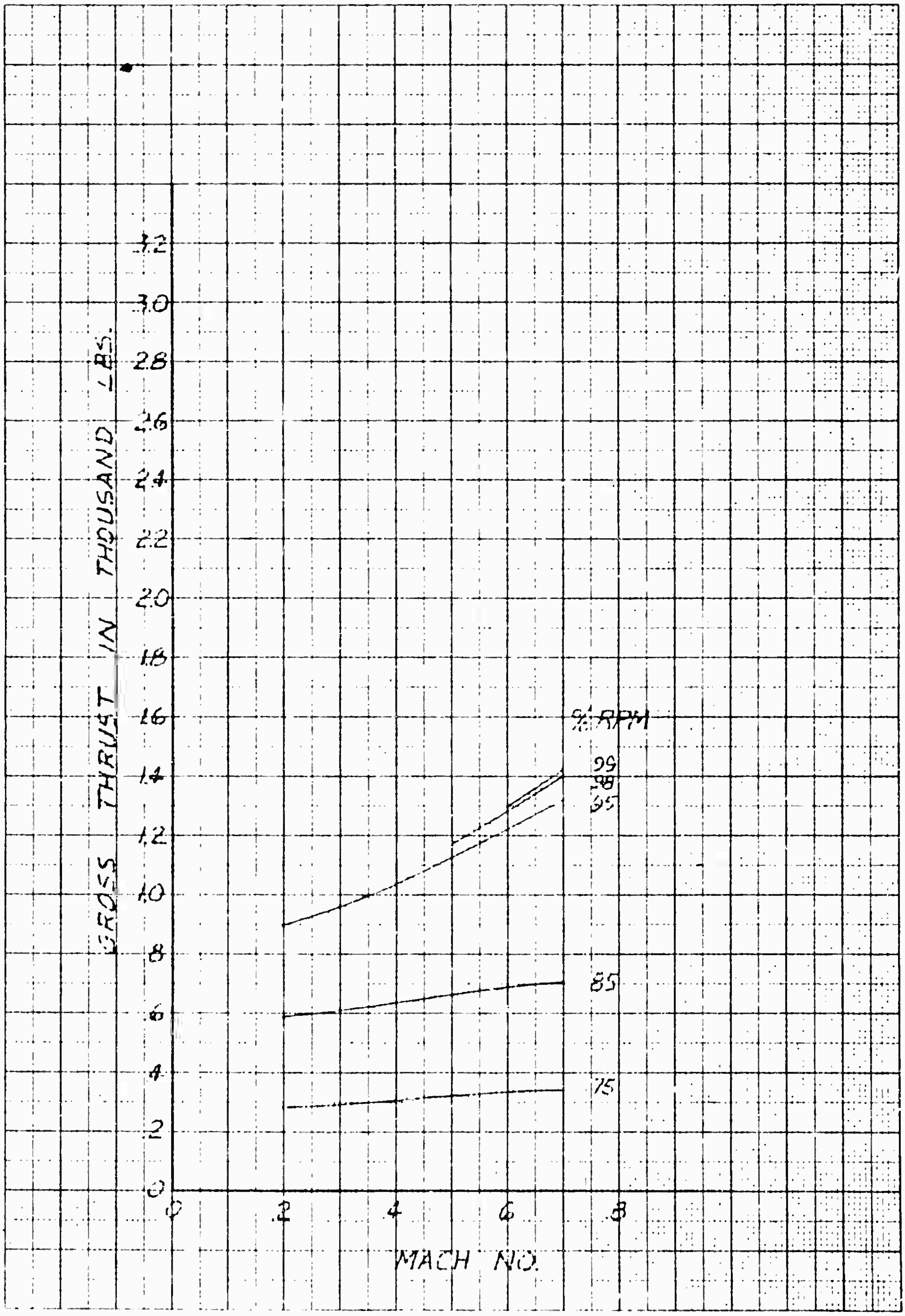

Figure 4.29 Gross Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Standard Day

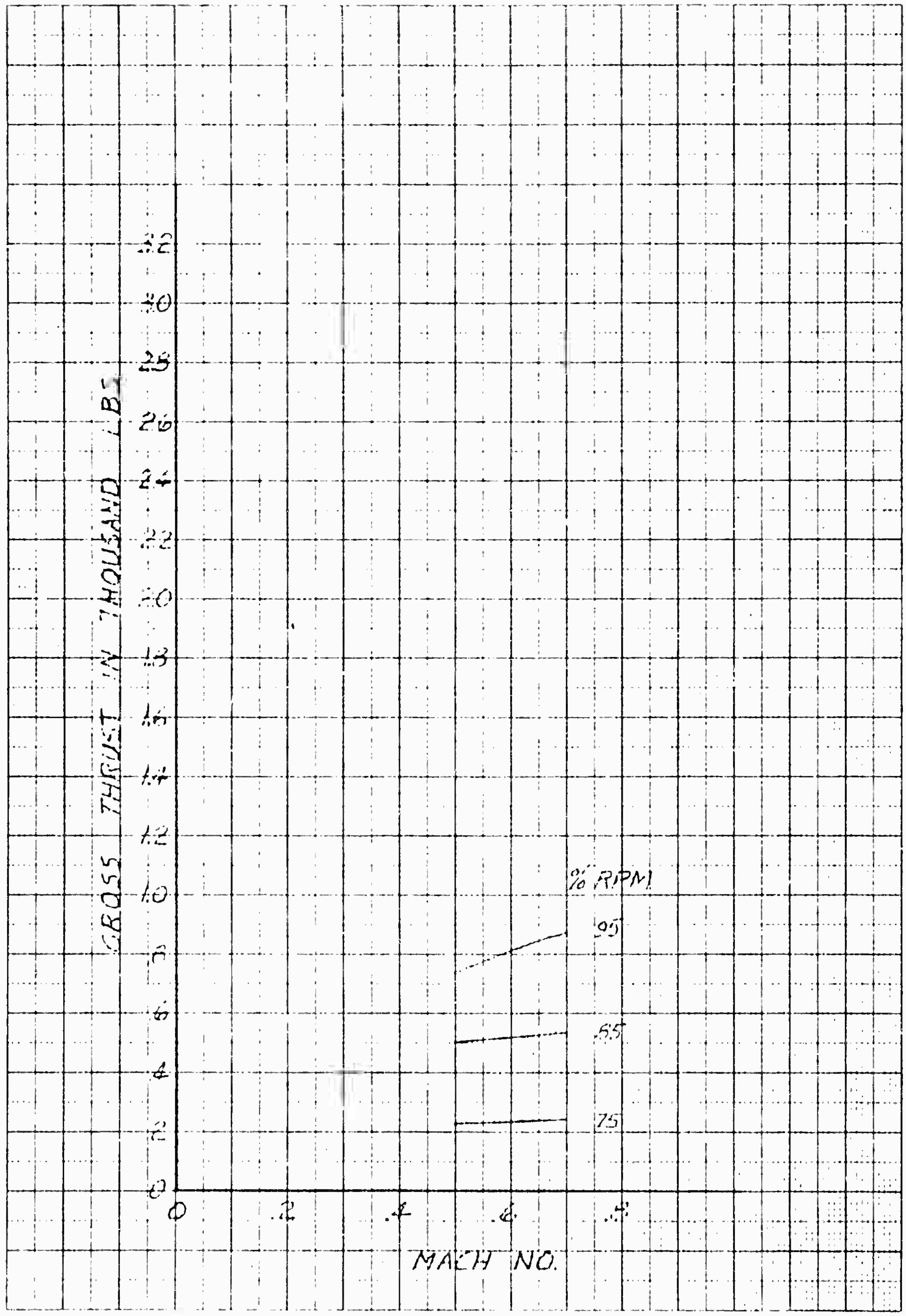

Figure 4.30 Gross Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Standard Day

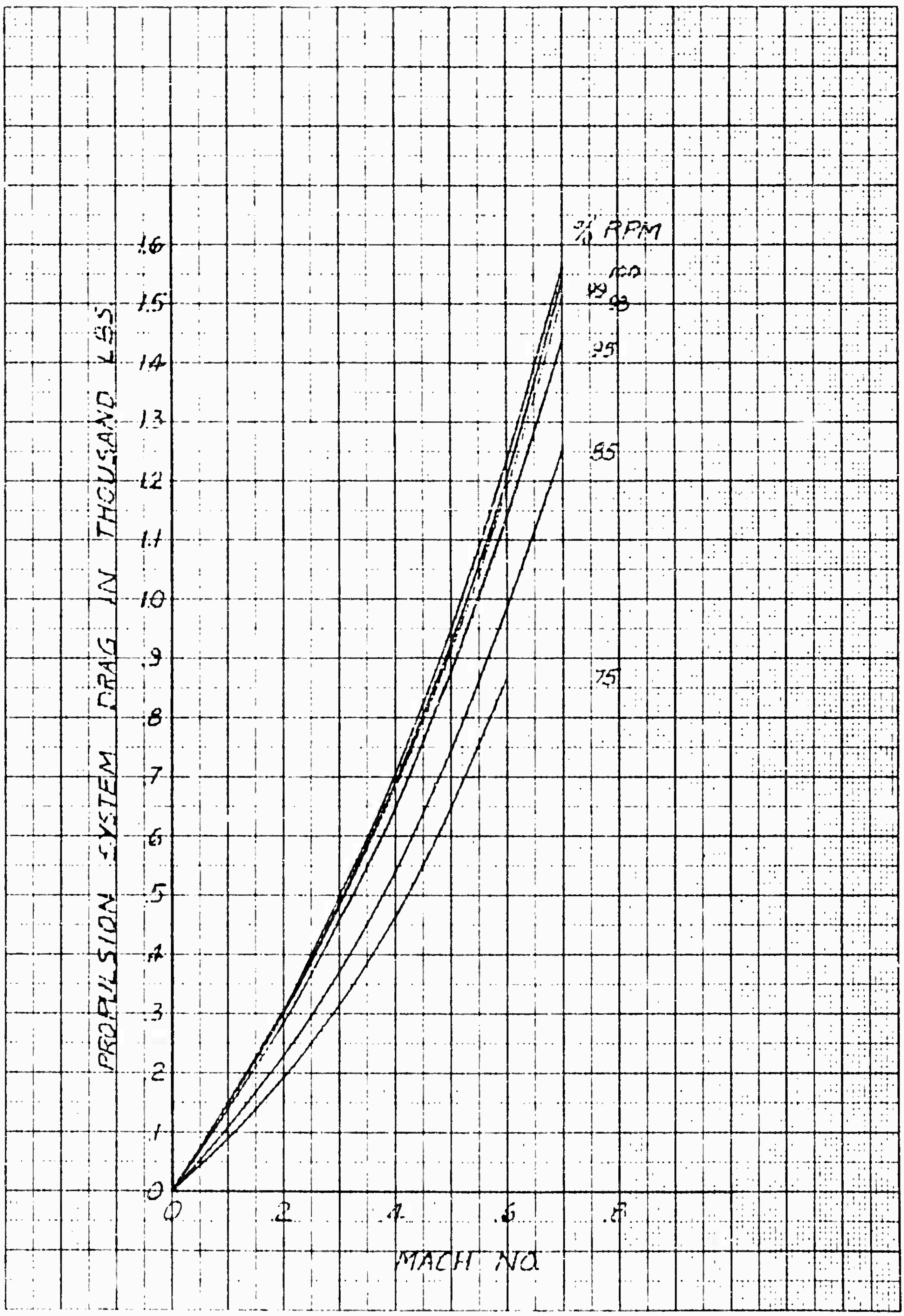

Figure 4.31 Propulsion System Drag vs Mach No. and % RPM;
Altitude = 0 ft., 1 Engine, Standard Day

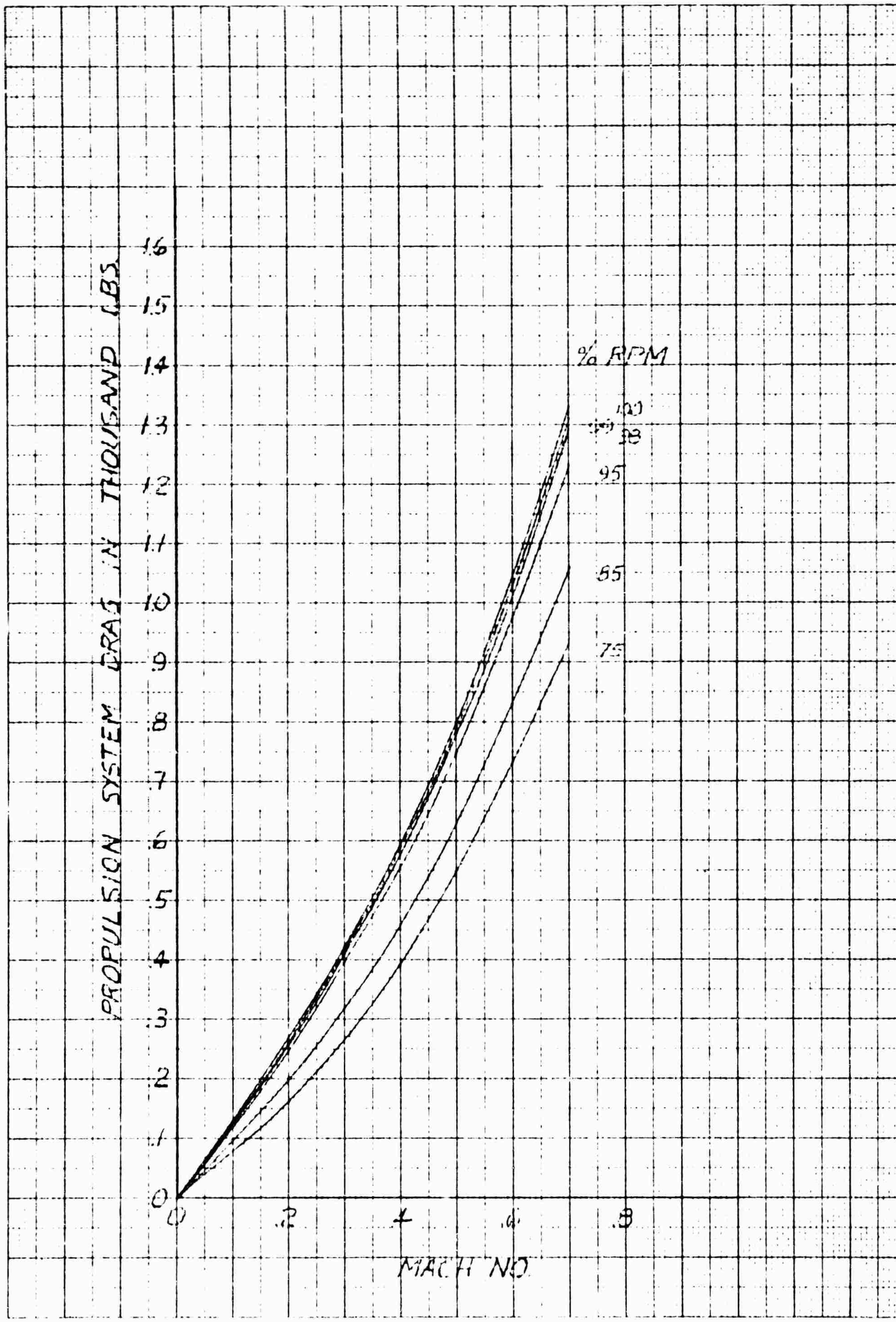

Figure 4.32 Propulsion System Drag vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Standard Day

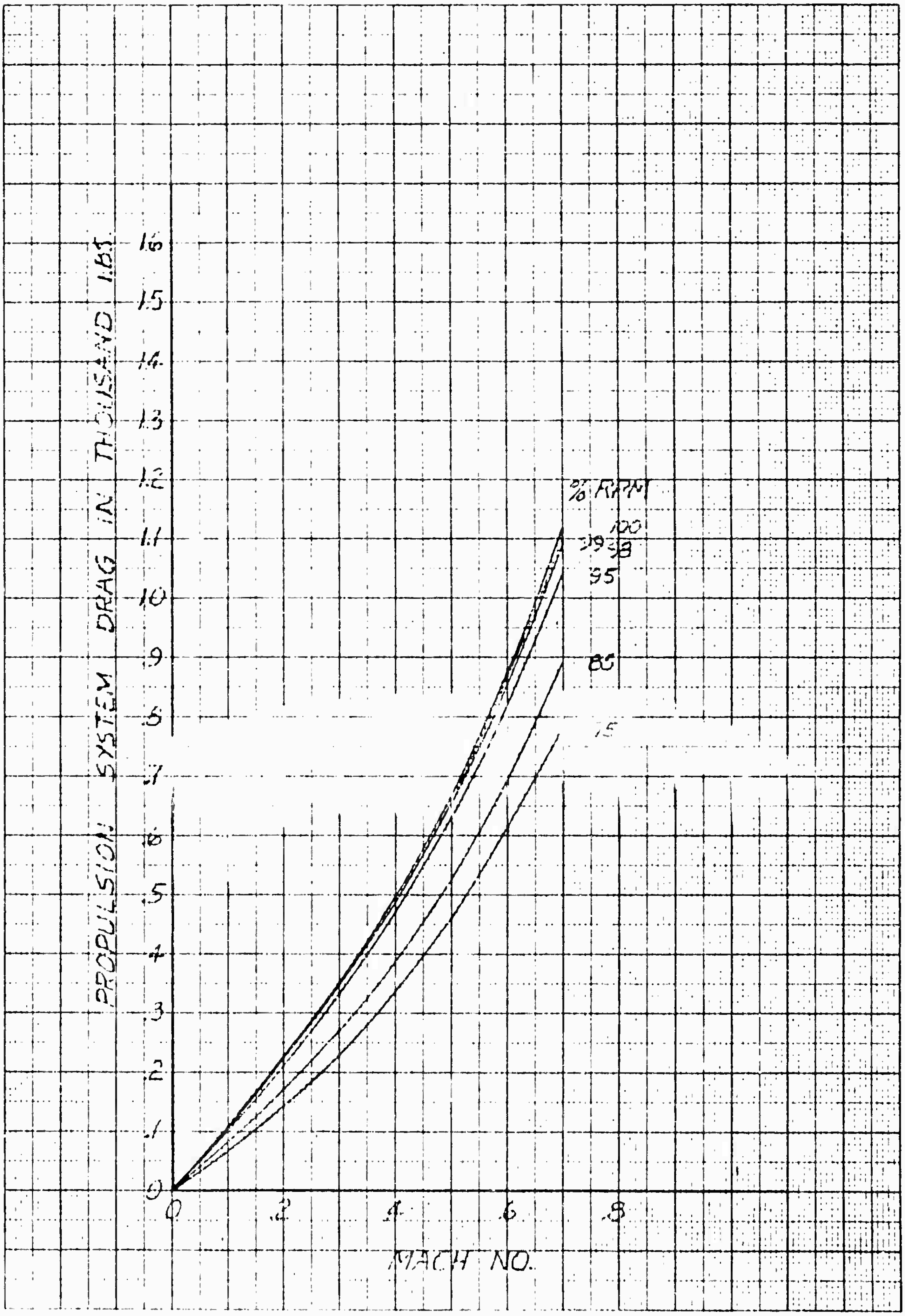

Figure 4.33 Propulsion System Drag vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Standard Day

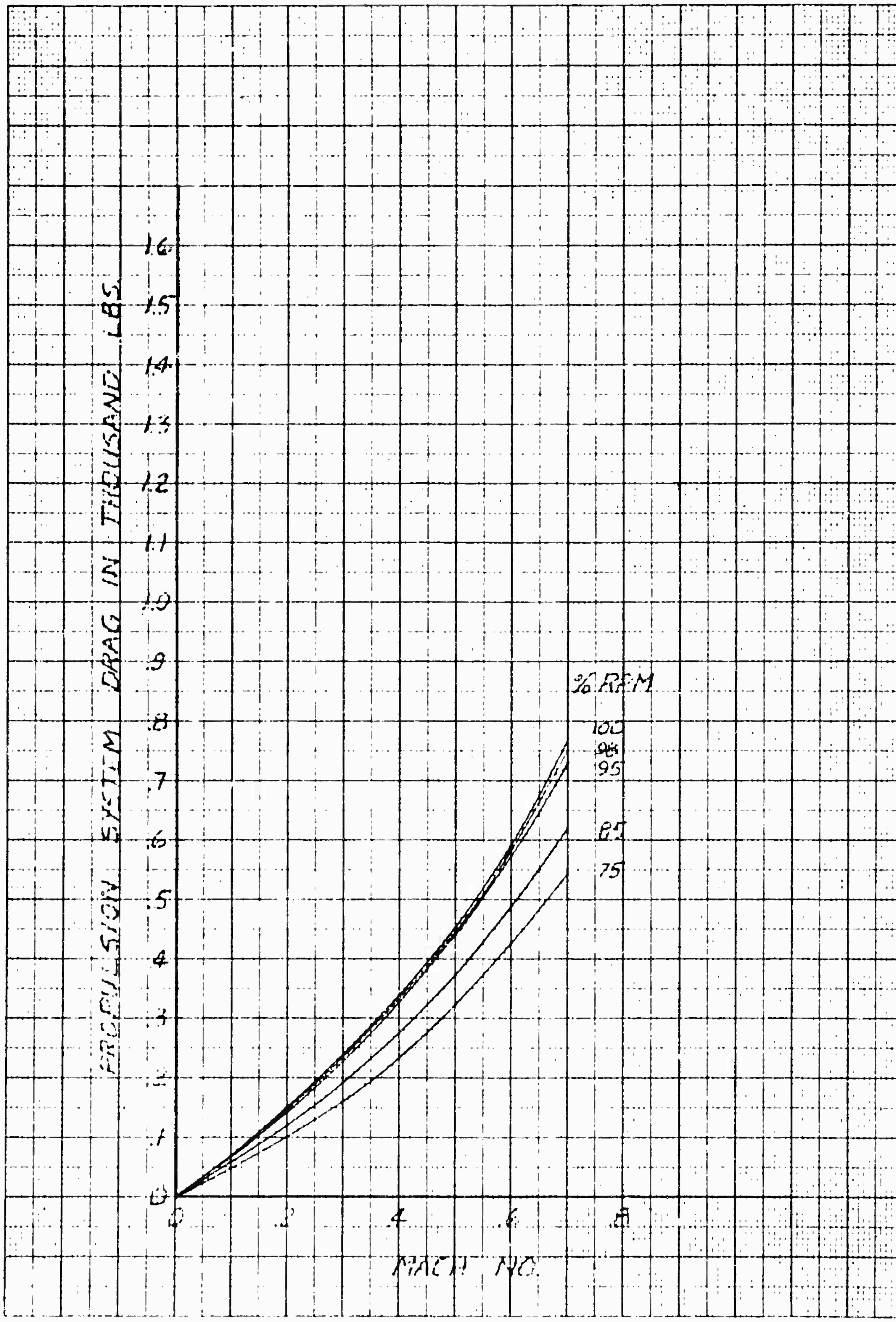

Figure 4.34 Propulsion System Drag vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Standard Day

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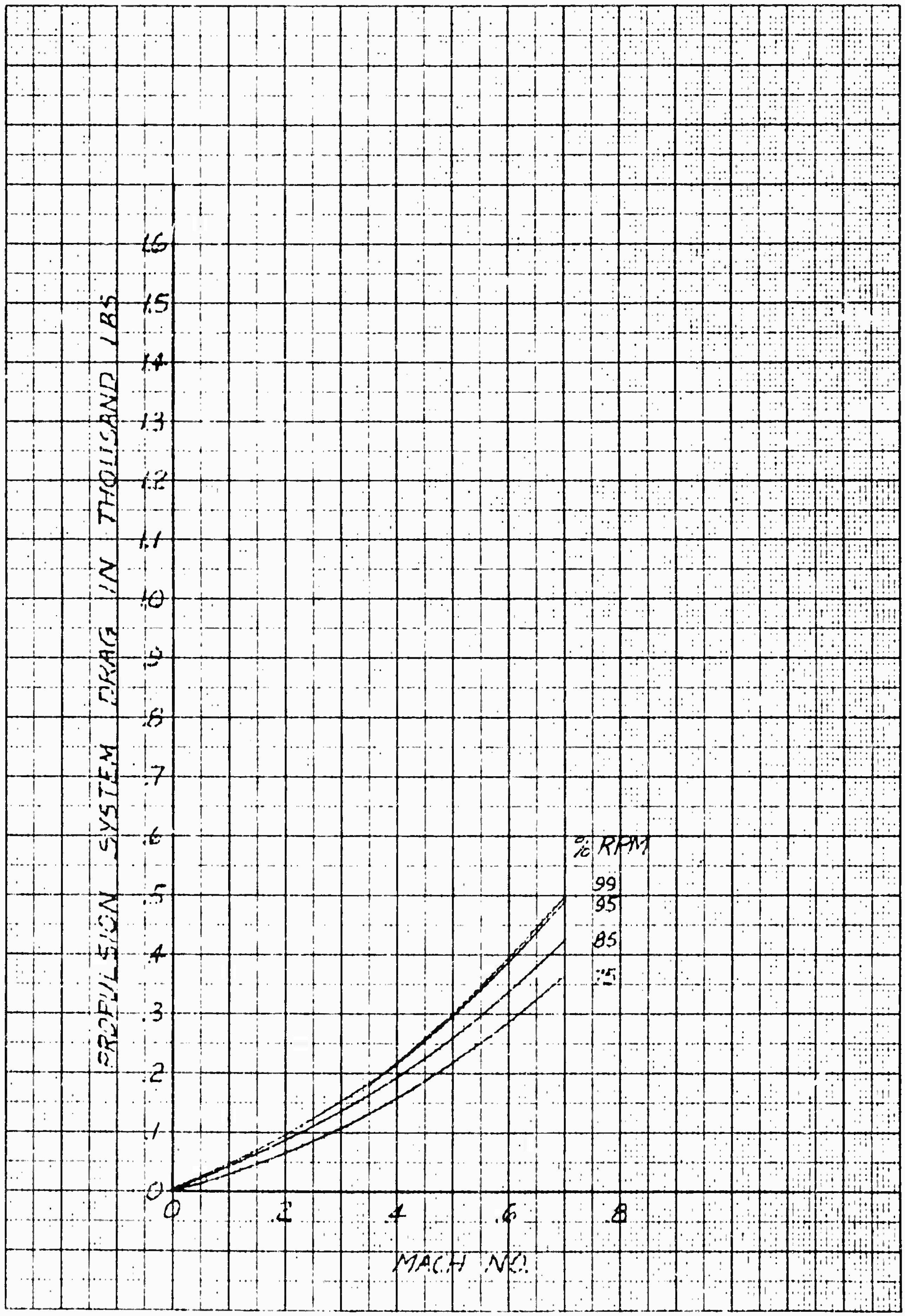

Figure 4.35 Propulsion System Drag vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Standard Day

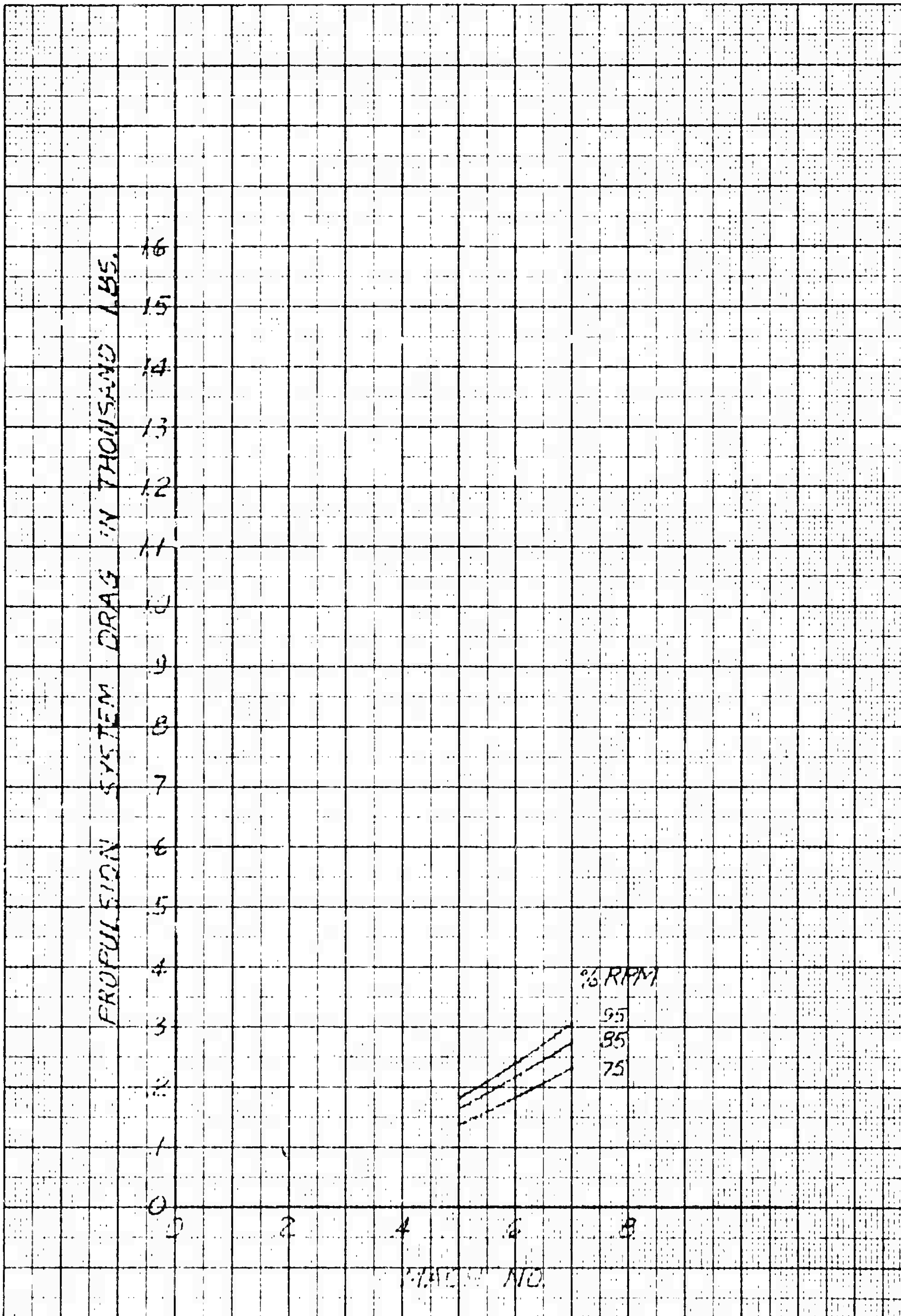

Figure 4.36 Propulsion System Drag vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Standard Day

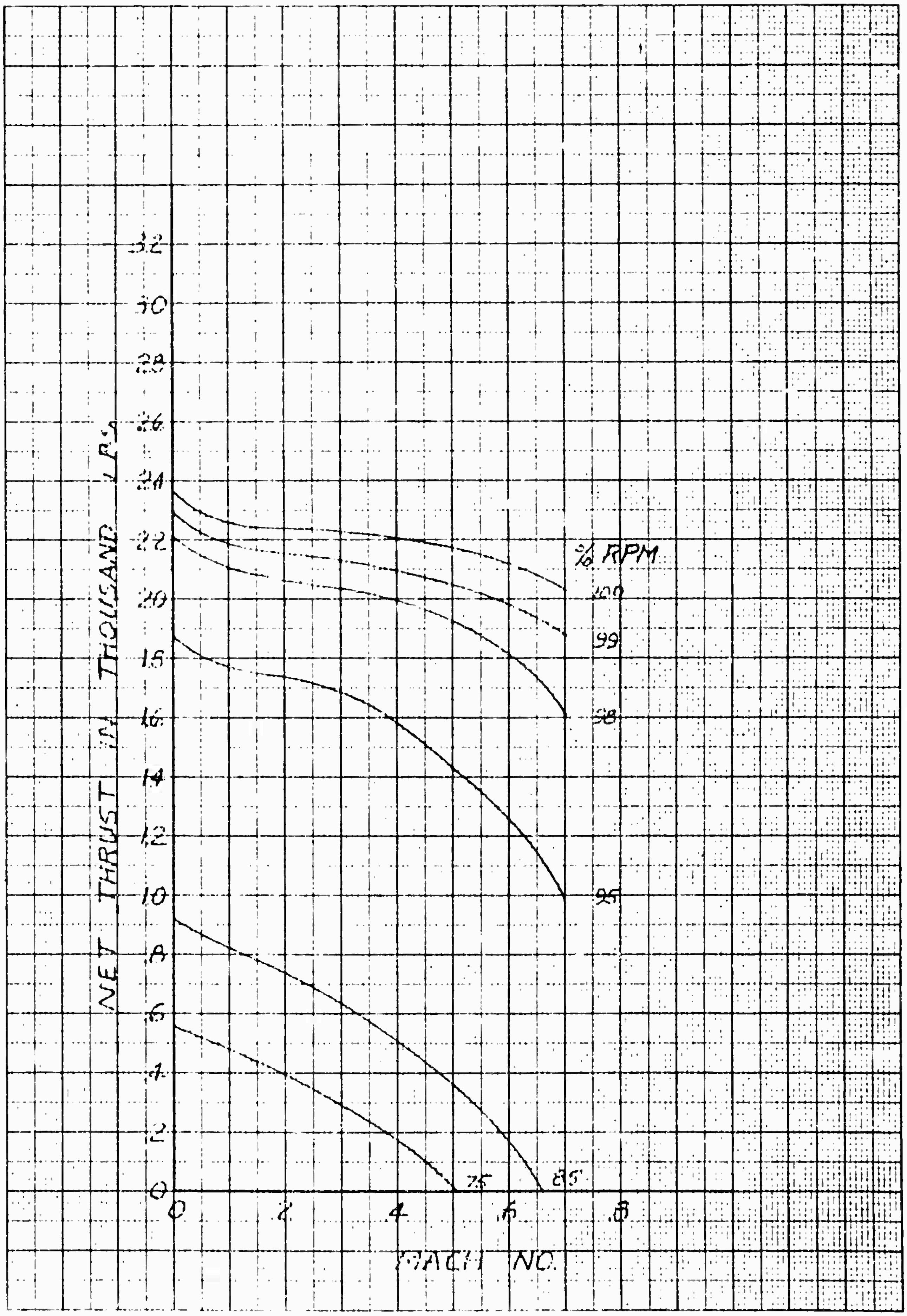

Figure 4.37 Net Thrust vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Standard Day

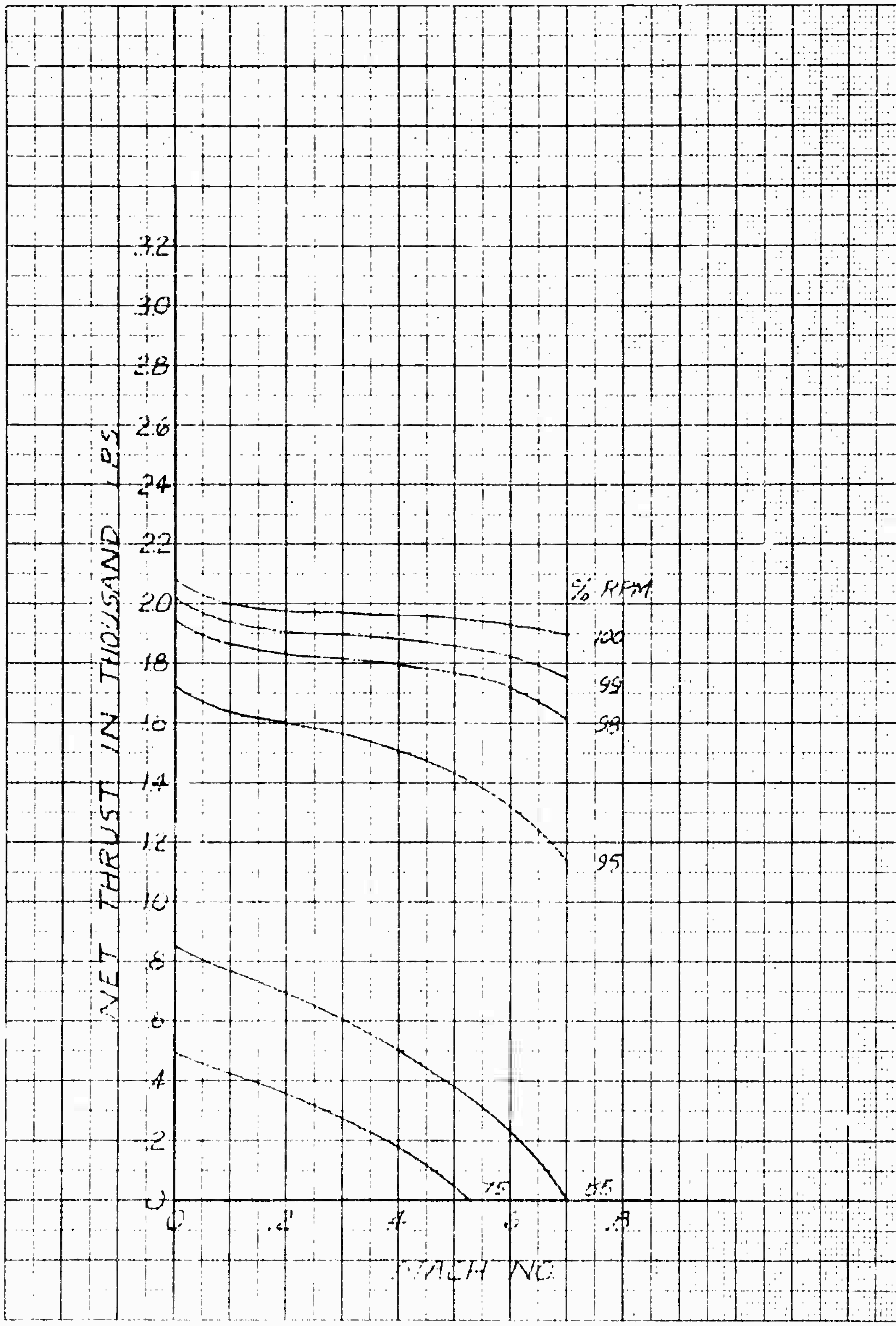

Figure 4.38 Net Thrust vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Standard Day

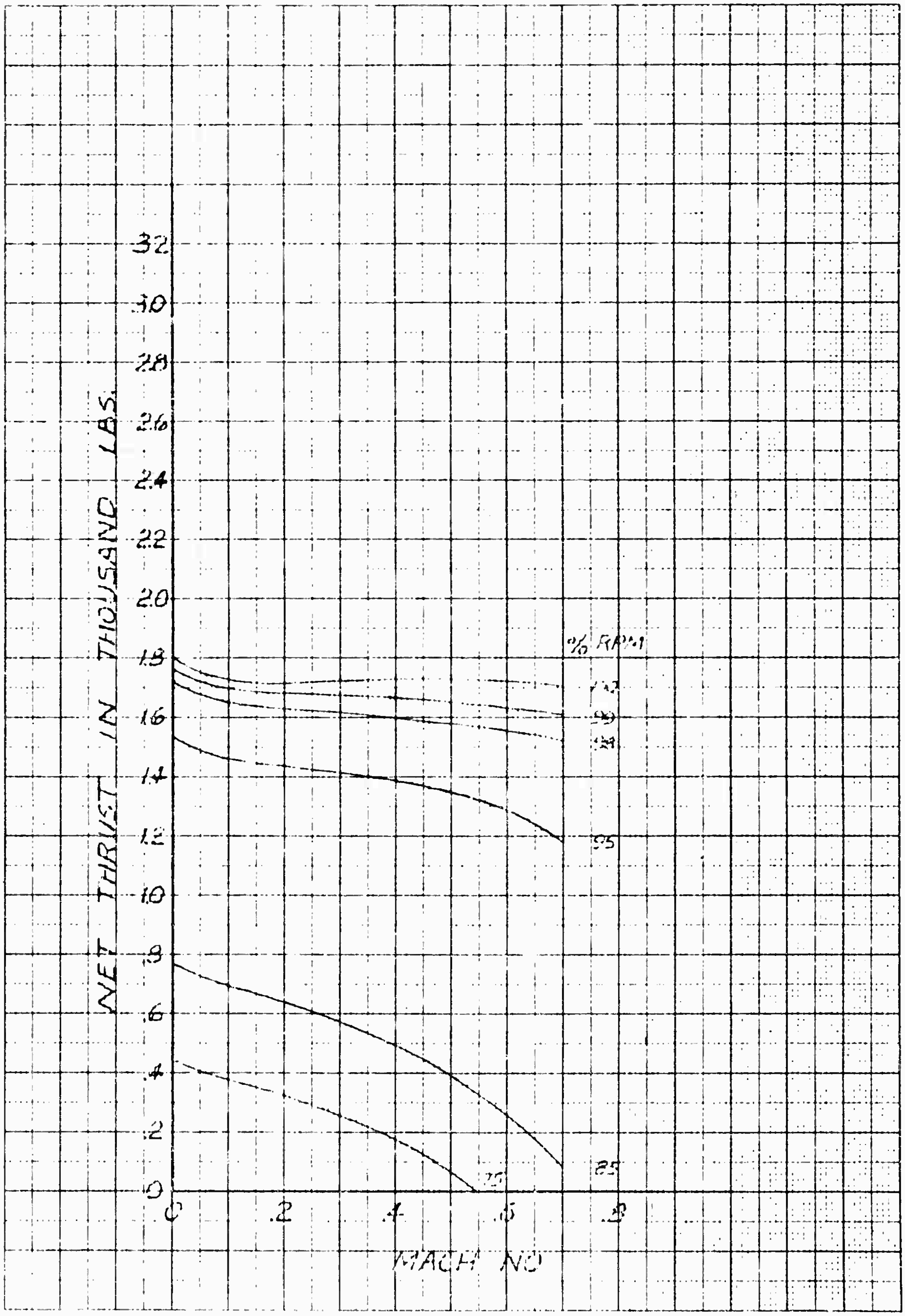

Figure 4.39 Net Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Standard Day

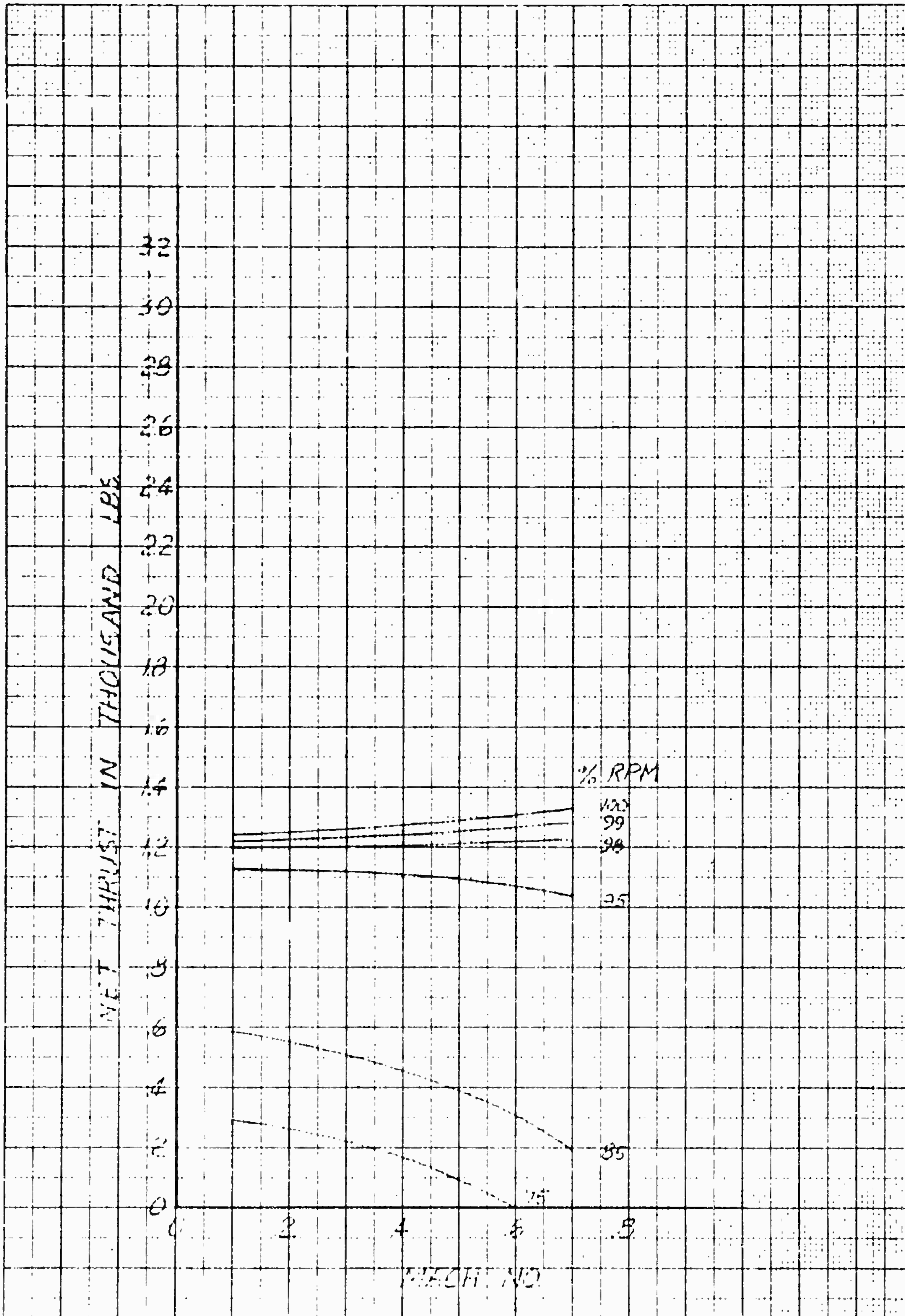

Figure 4.40 Net Thrust vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Standard Day

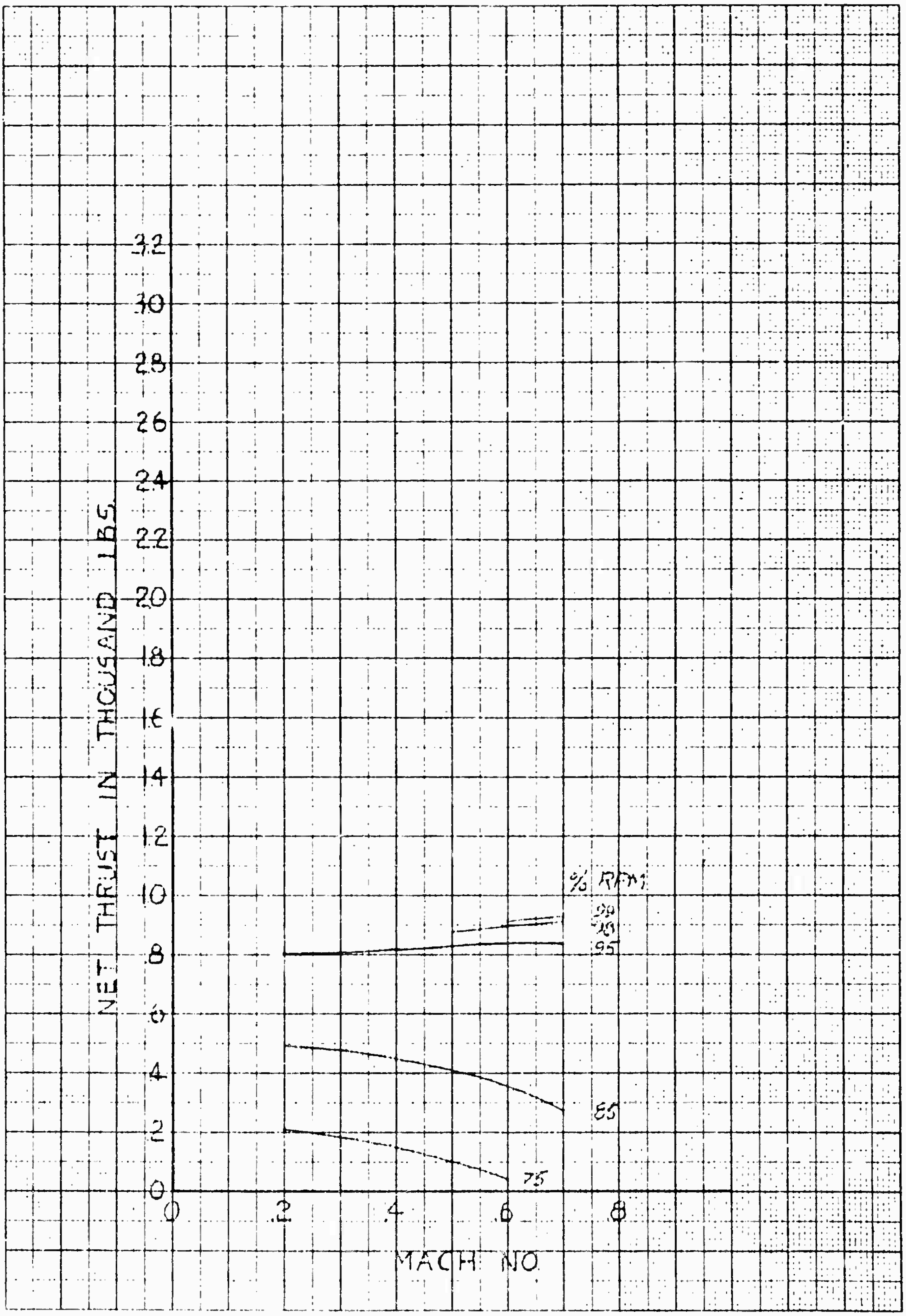

Figure 4.41 Net Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Standard Day

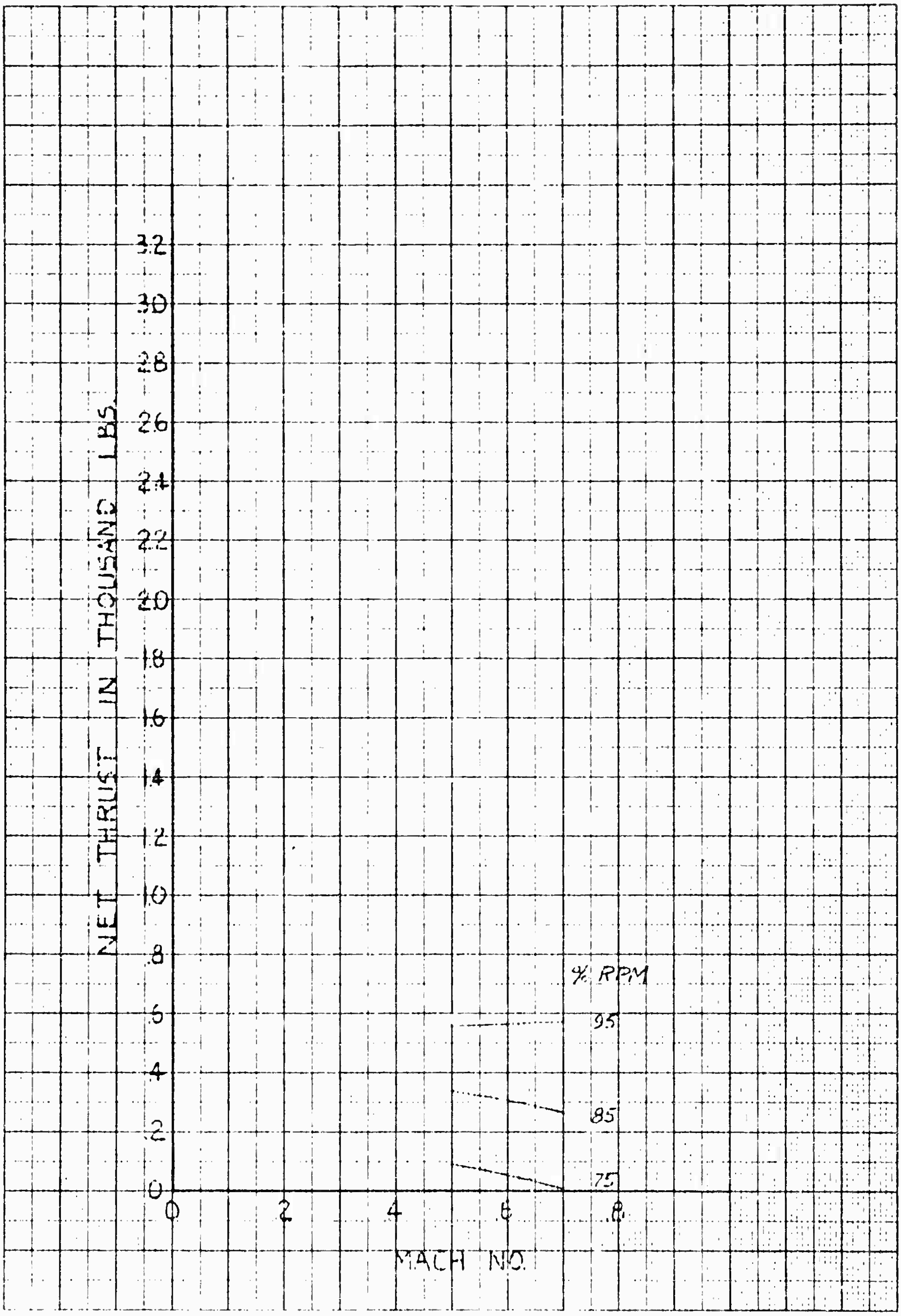

Figure 4.42 Net Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Standard Day

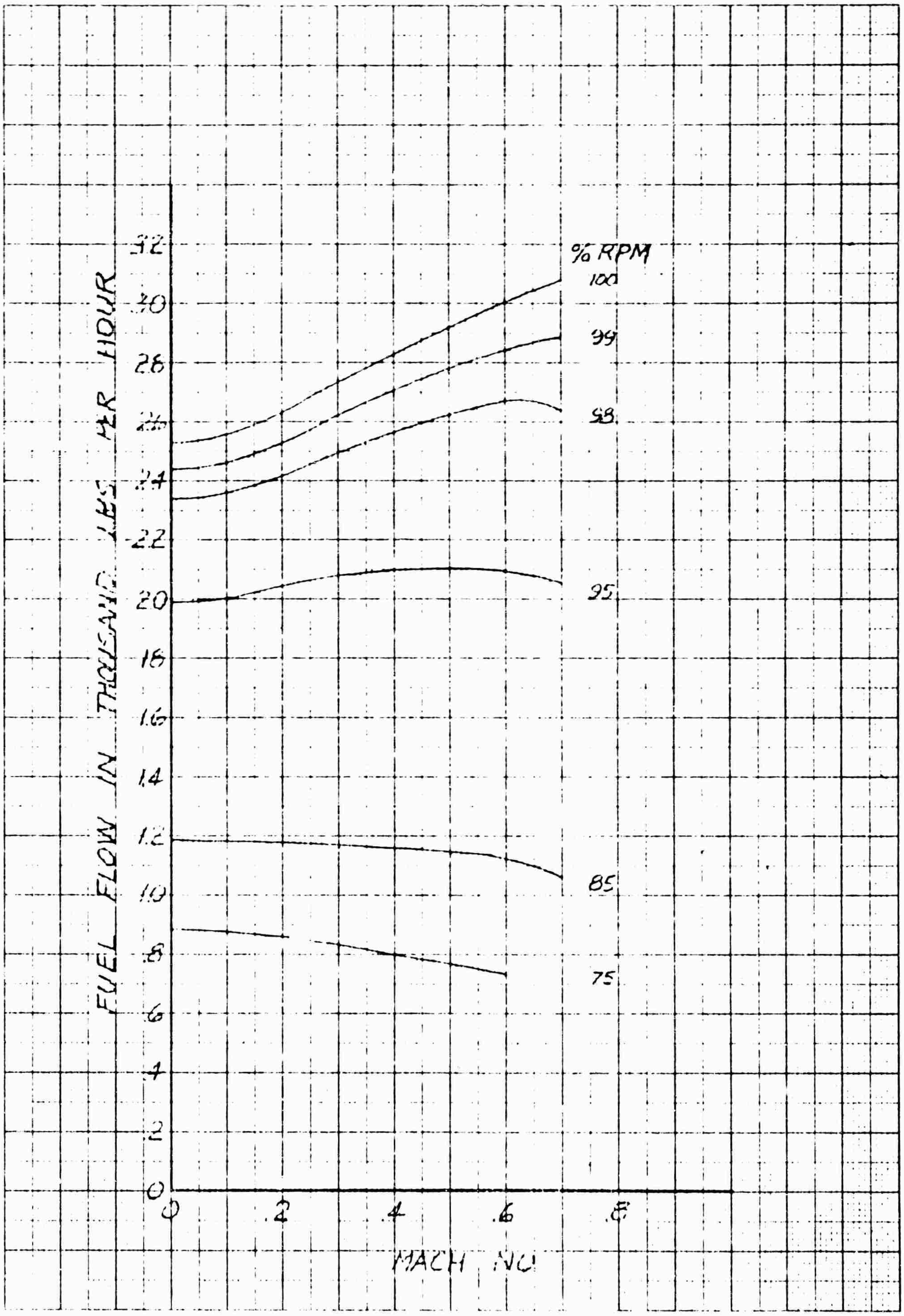

Figure 4.43 Fuel Flow vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Standard Day

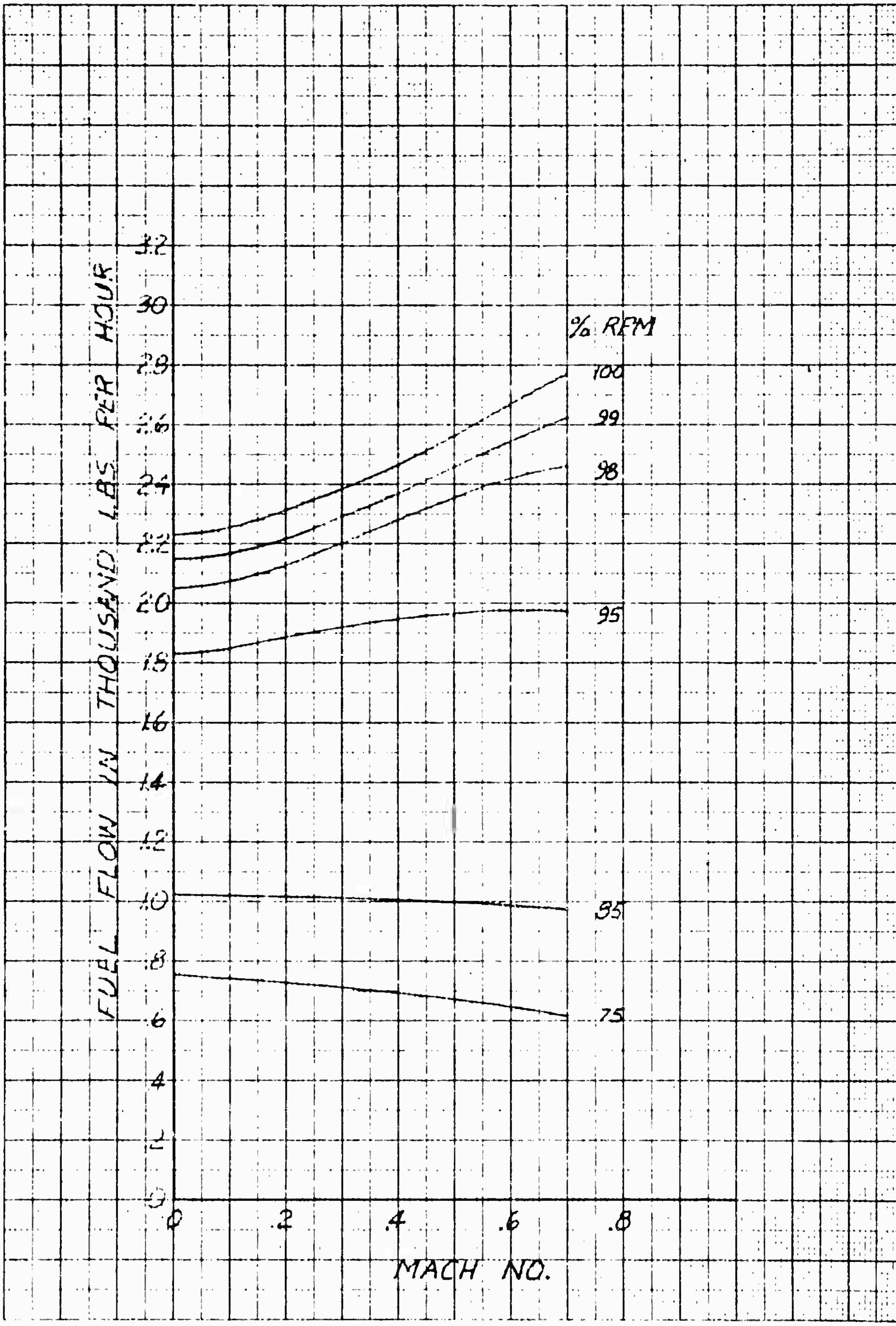

Figure 4.44 Fuel Flow vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Standard Day

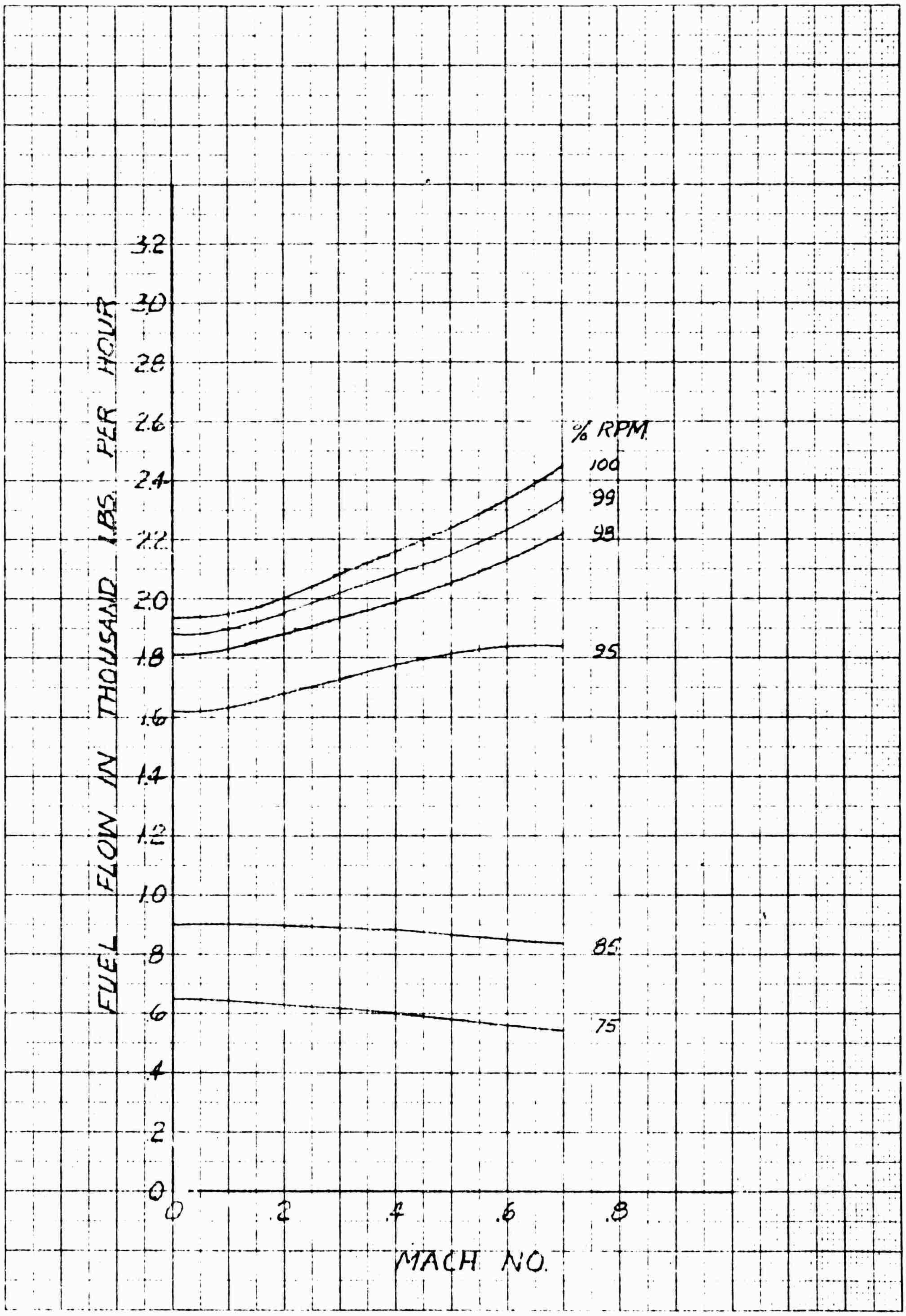

Figure 4.45 Fuel Flow vs. Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Standard Day

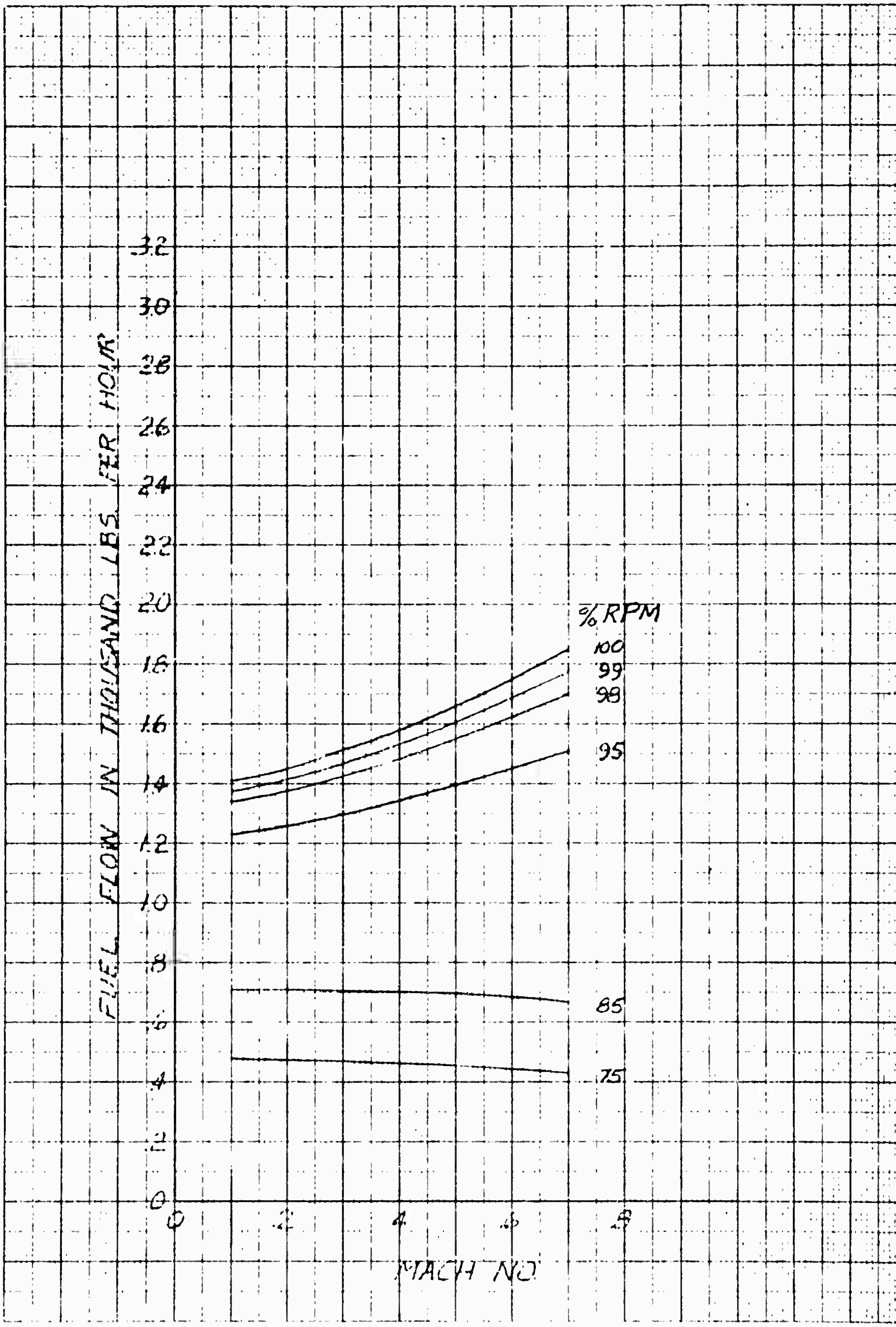

Figure 4.46 Fuel Flow vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Standard Day

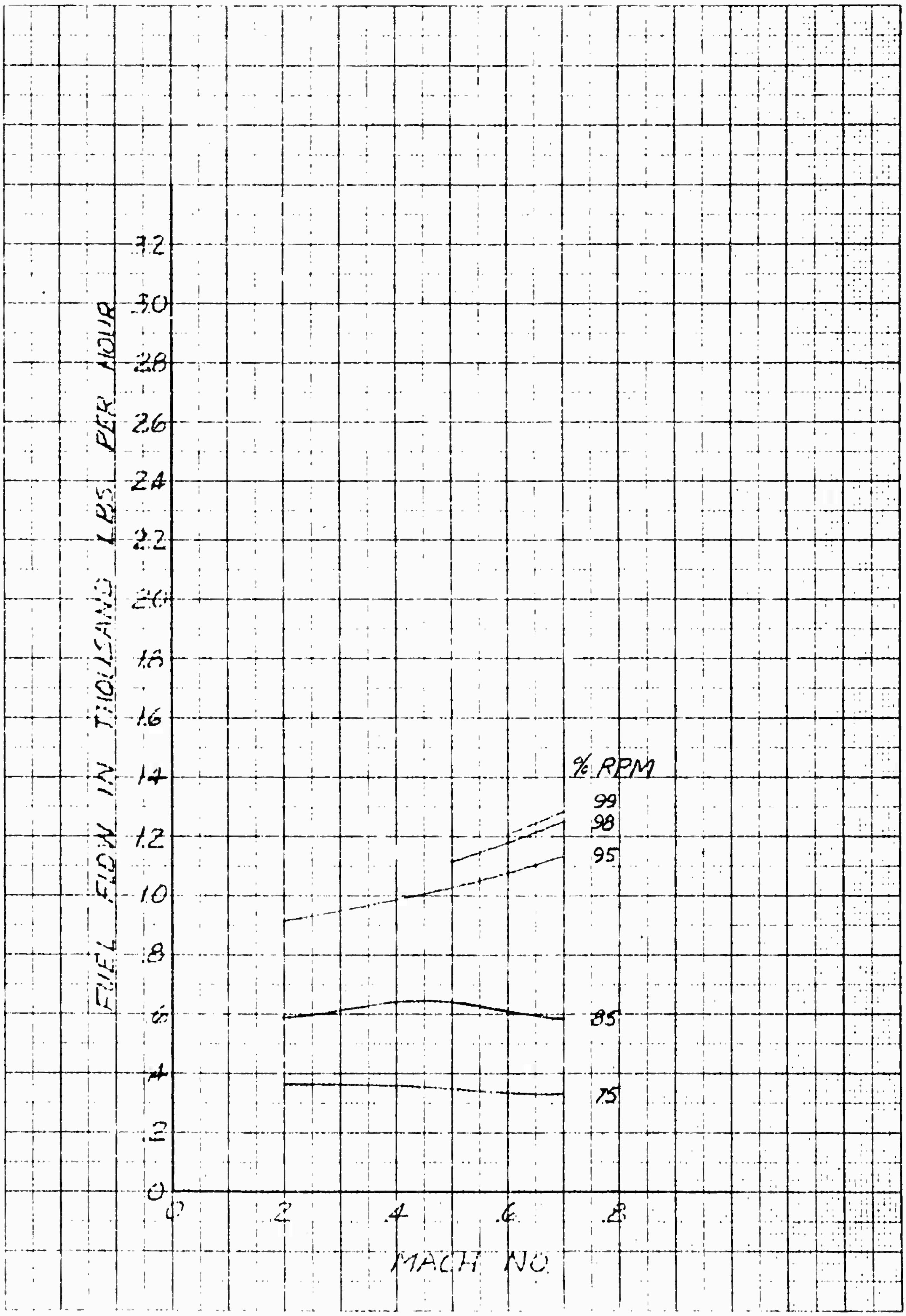

Figure 4.47 Fuel Flow vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Standard Day

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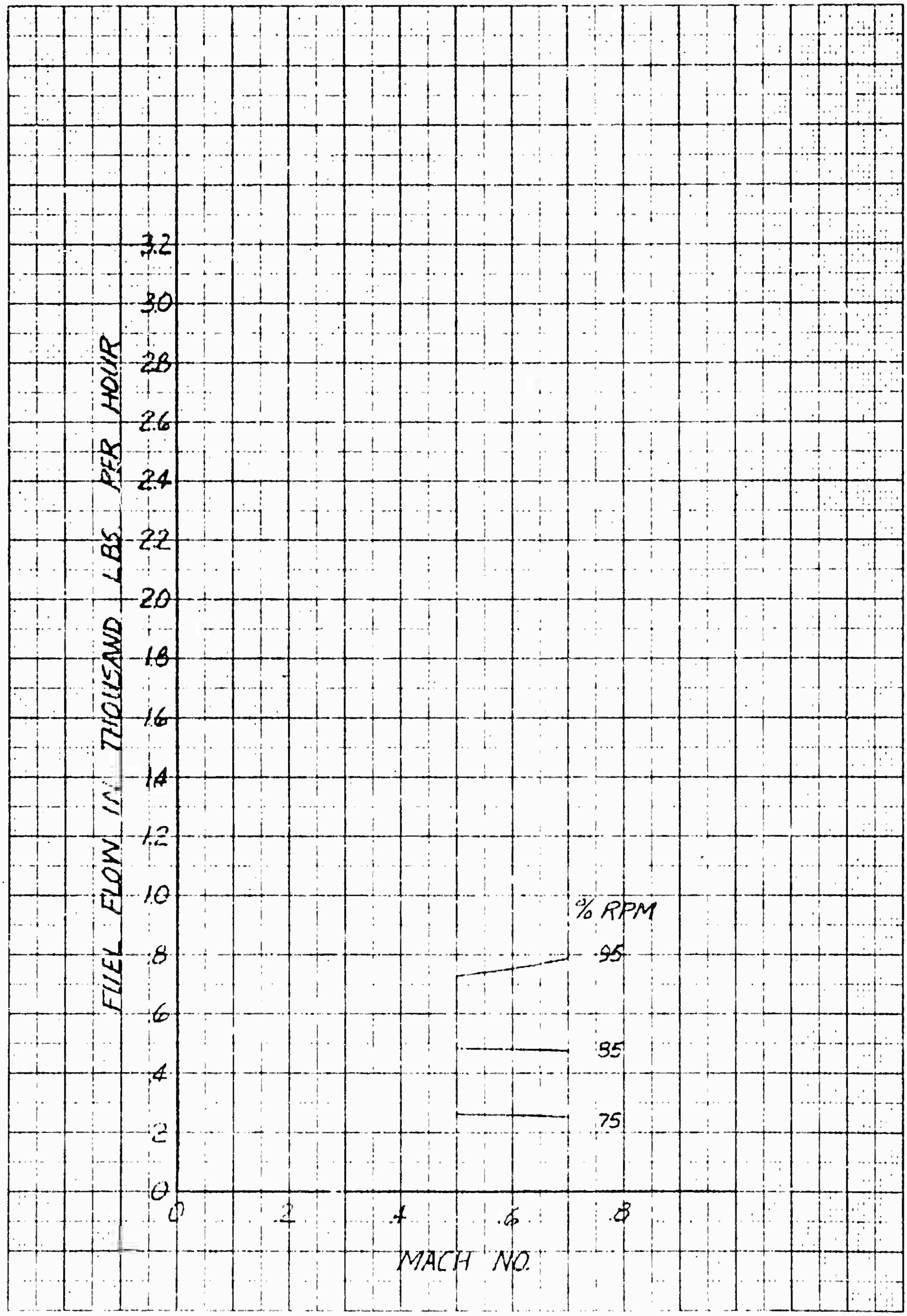

Figure 4.48 Fuel Flow vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Standard Day

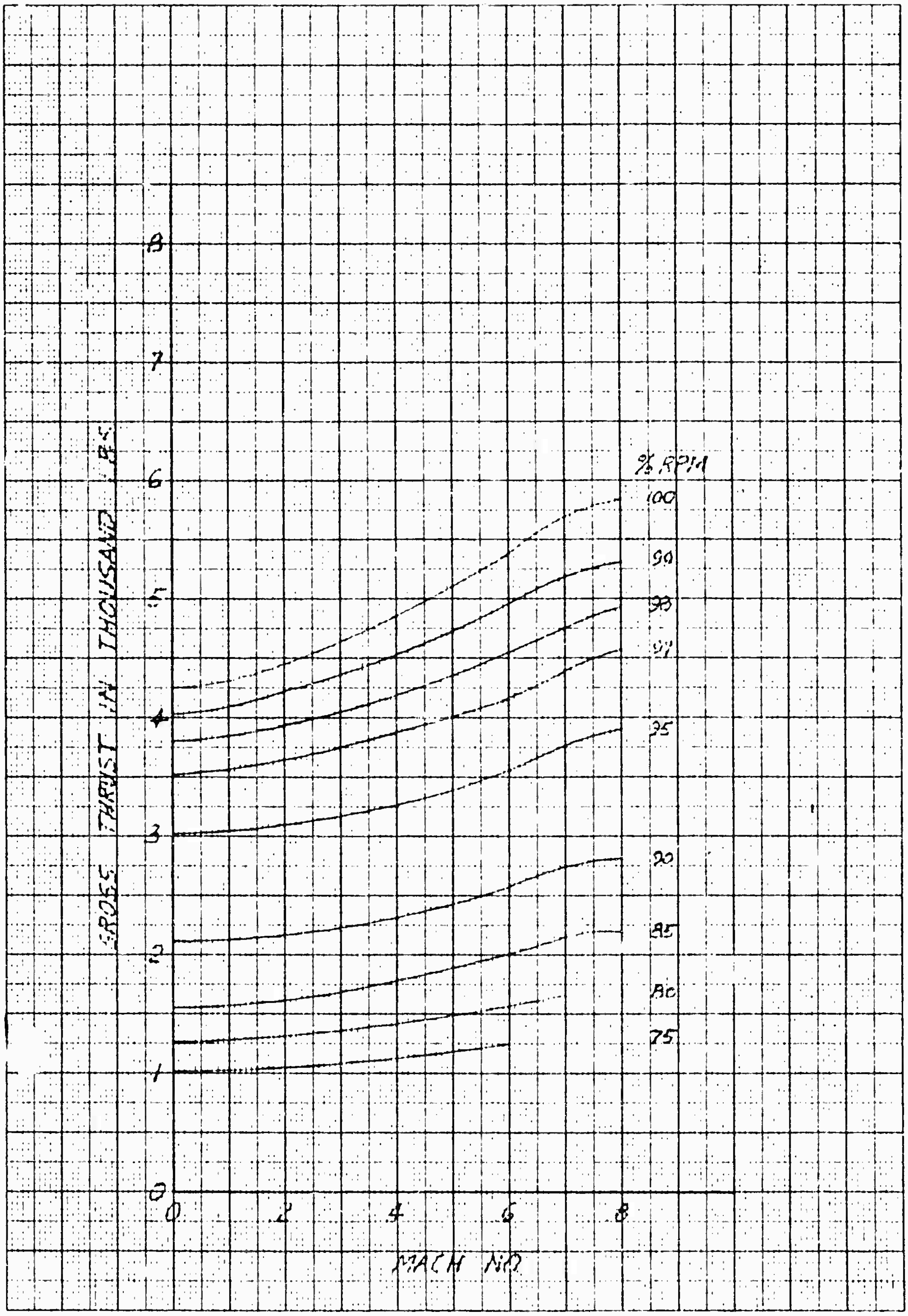

Figure 4.49 Gross Thrust vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Hot Day

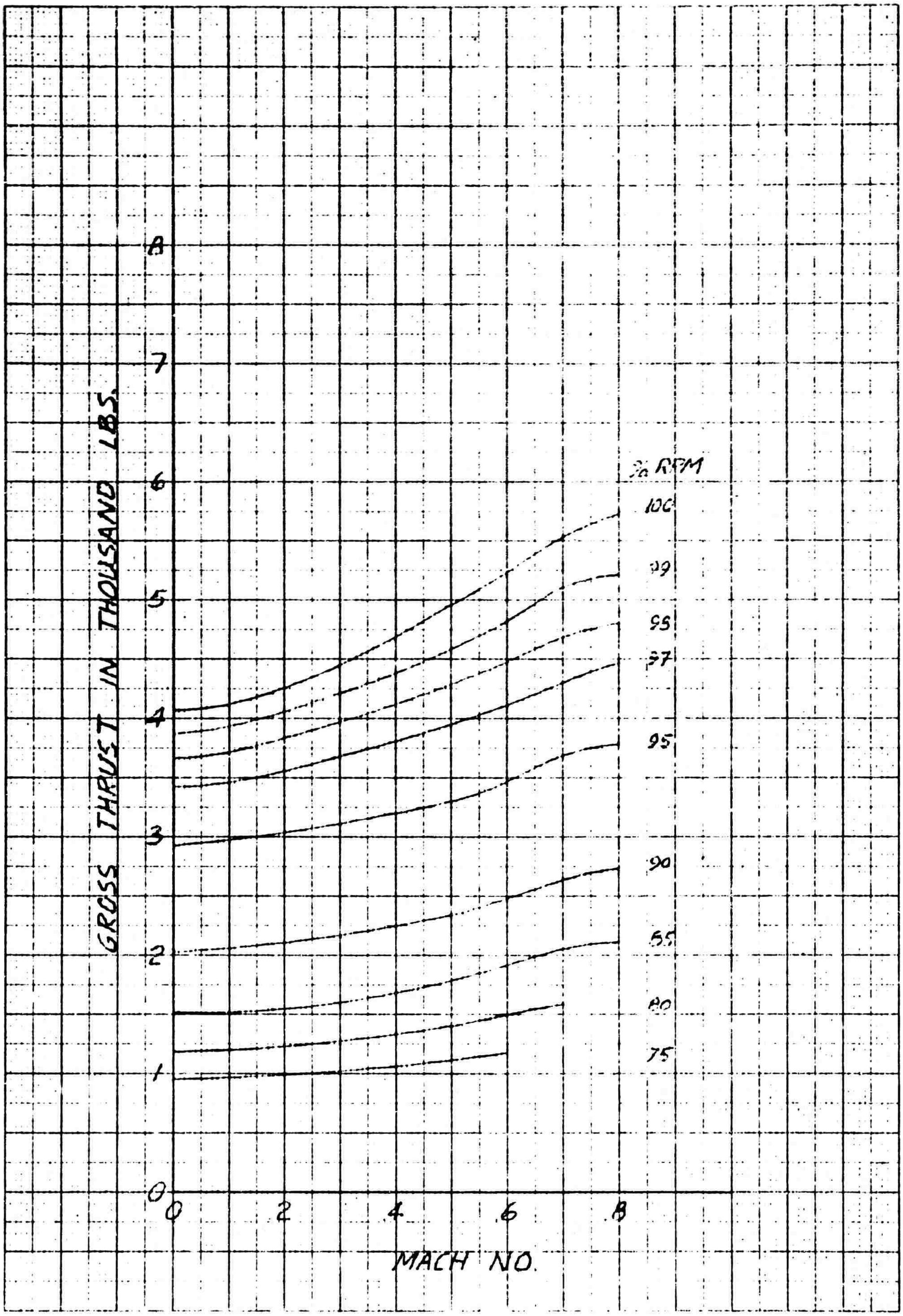

Figure 4.50 Gross Thrust vs Mach No. and % RPM; Altitude = 2500 ft., 2 Engines, Hot Day

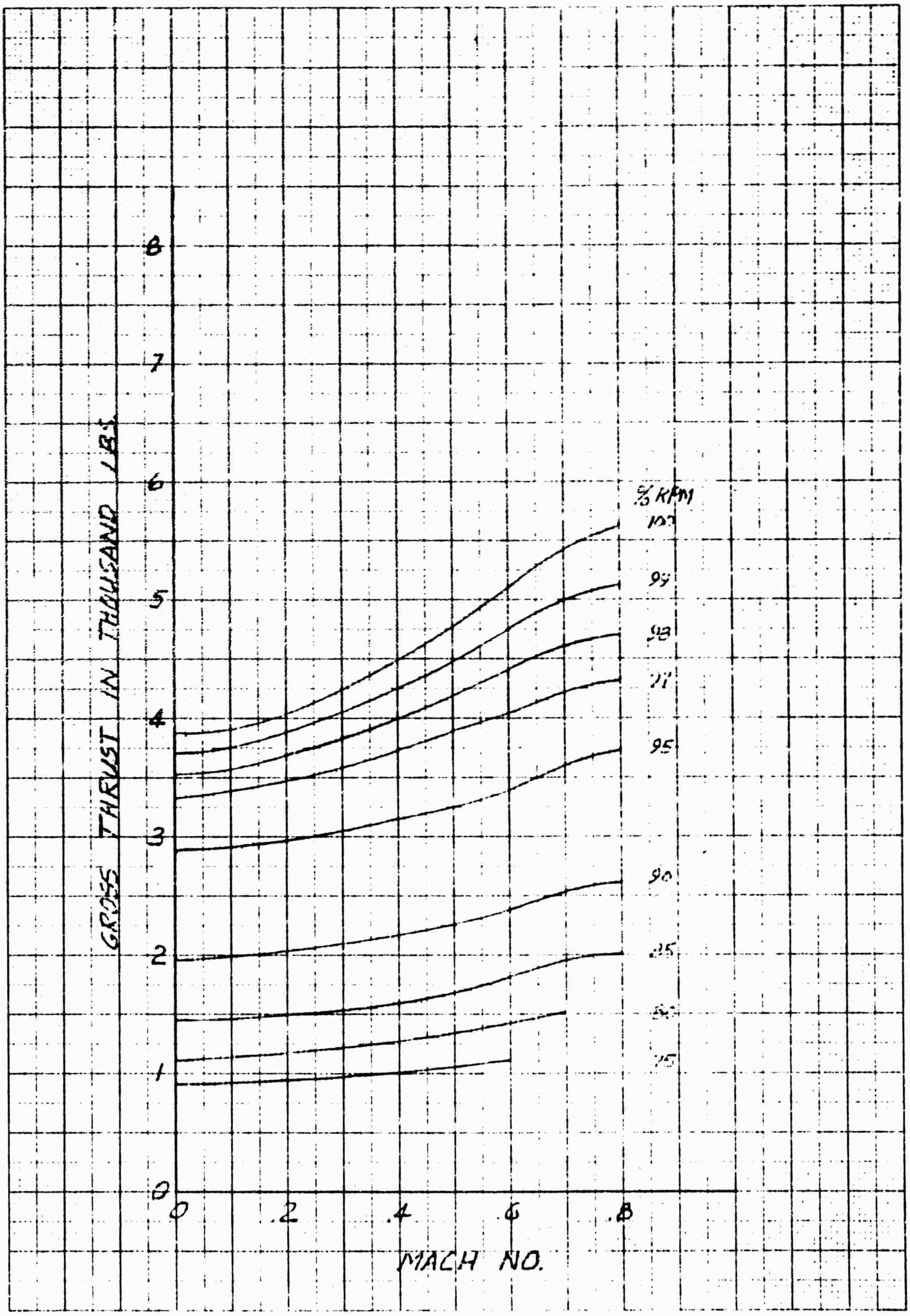

Figure 4.51 Gross Thrust vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Hot Day

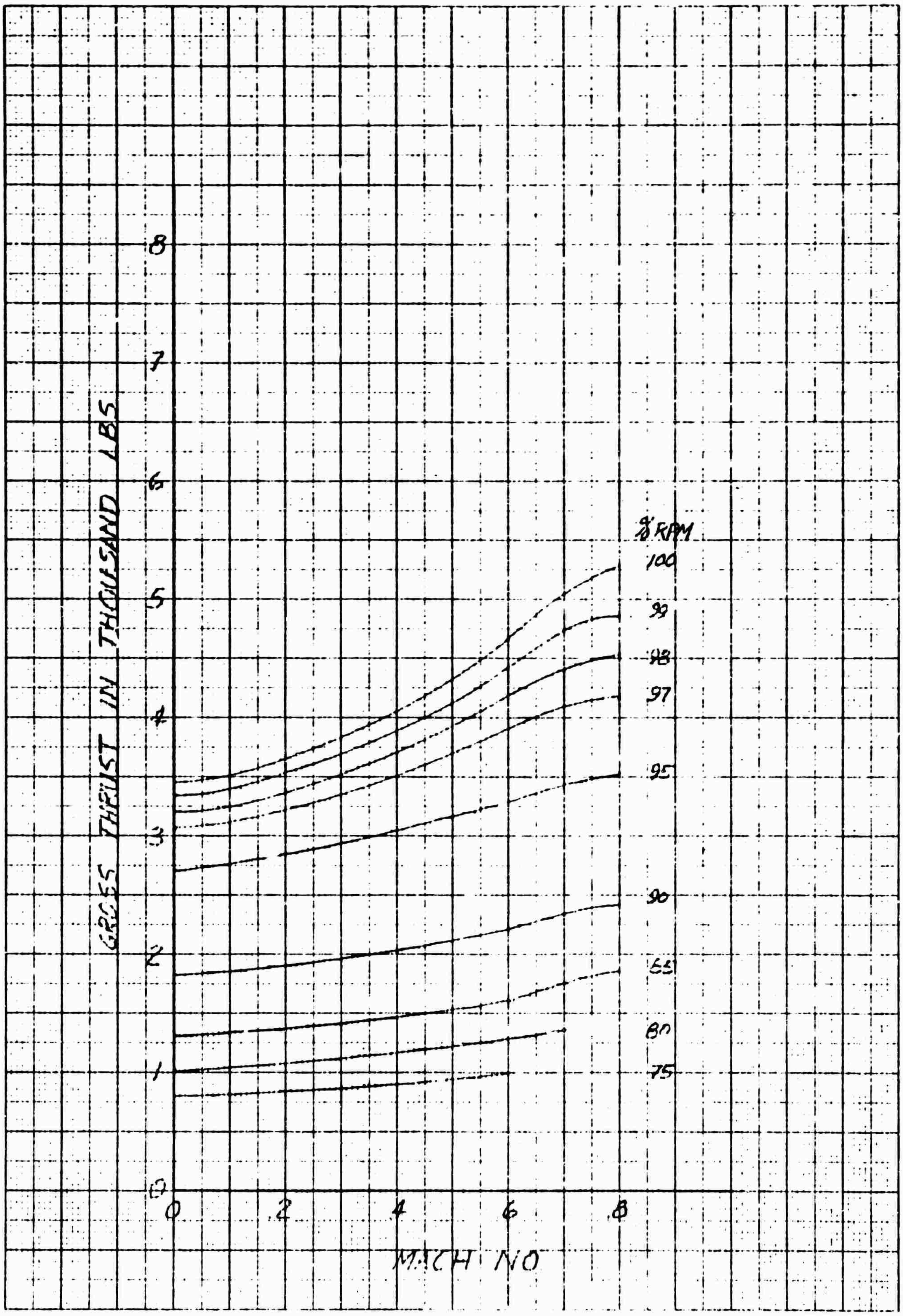

Figure 4.52 Gross Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Hot Day

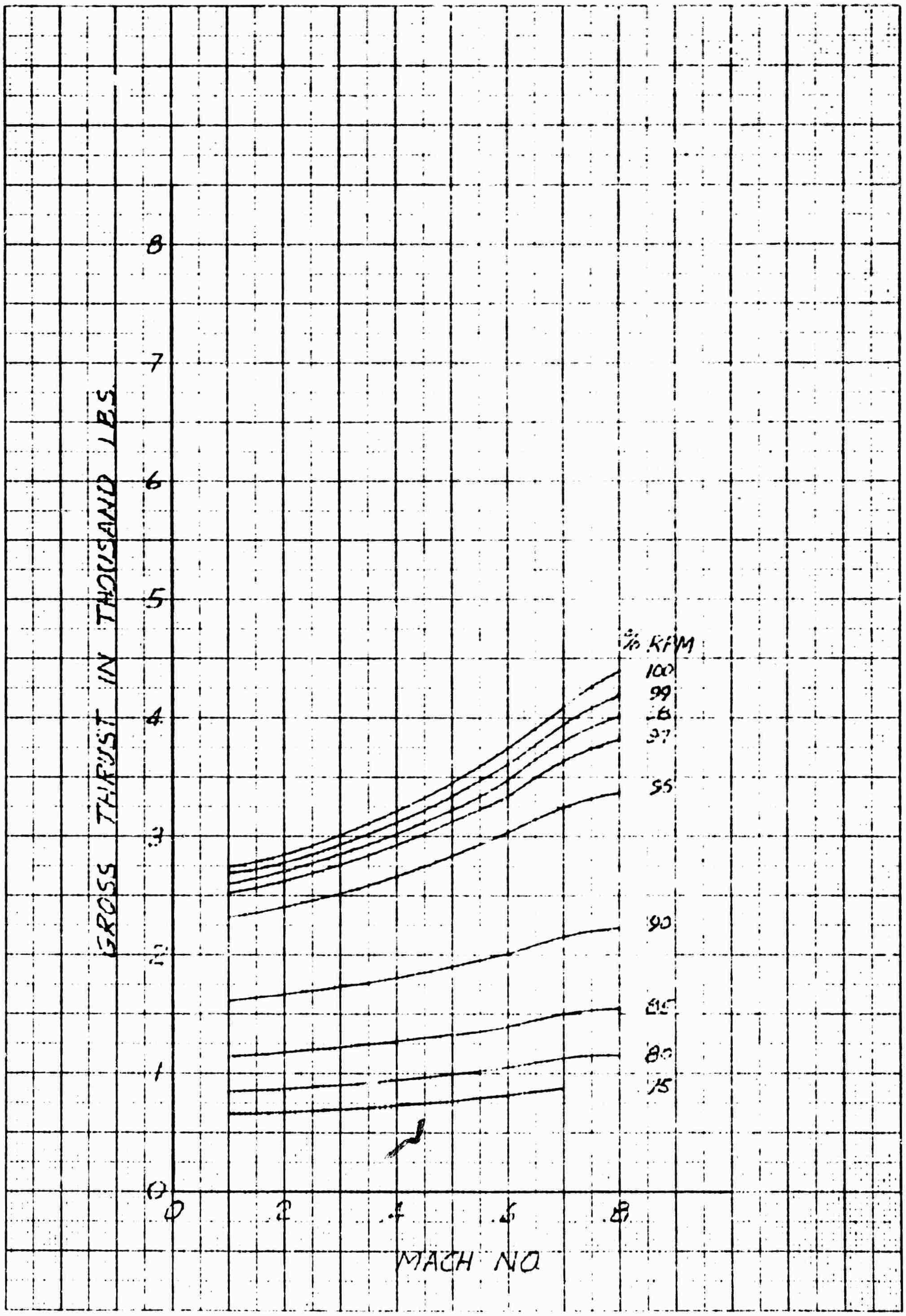

Figure 4.53 Gross Thrust vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Hot Day

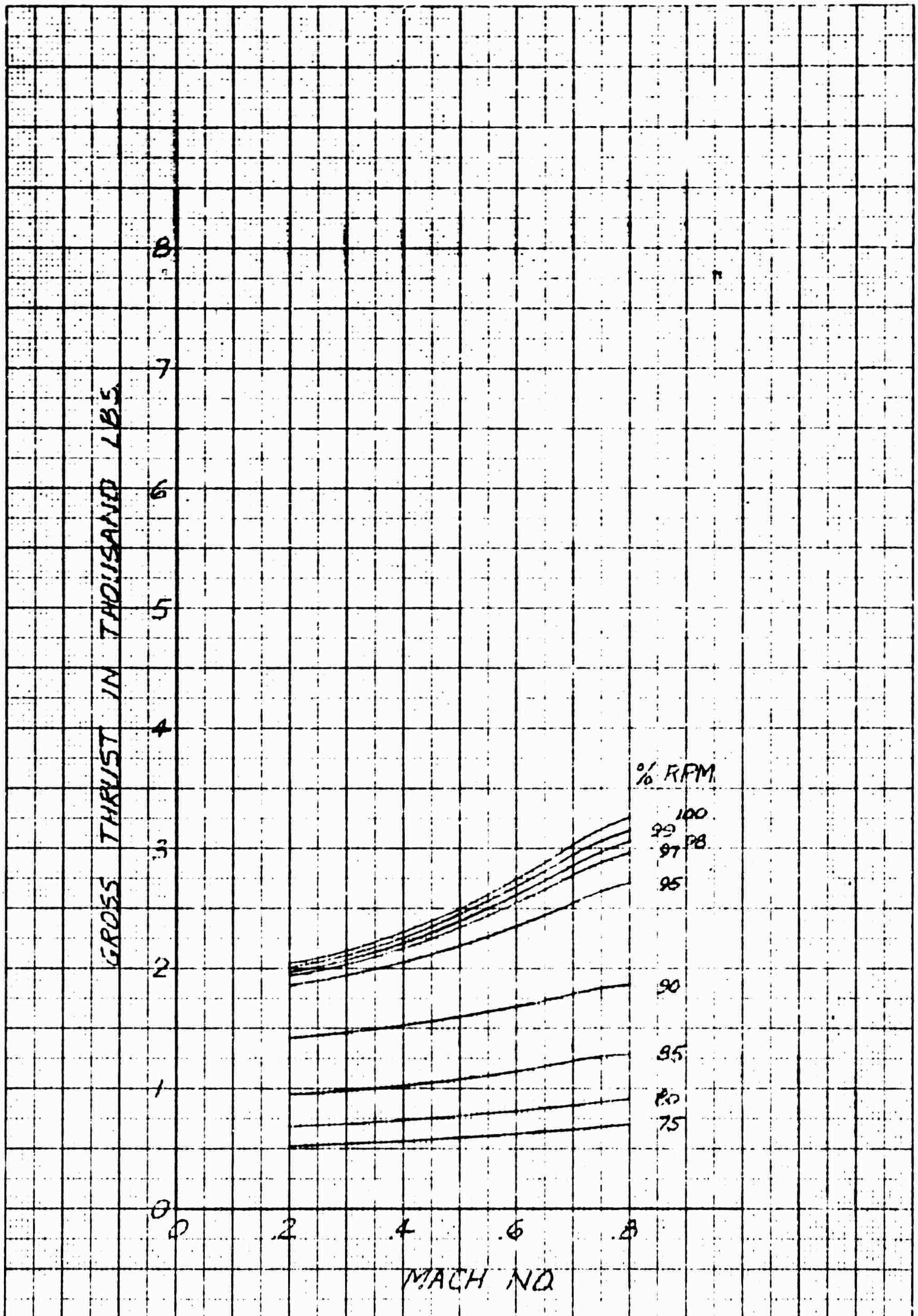

Figure 4.54 Gross Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Hot Day

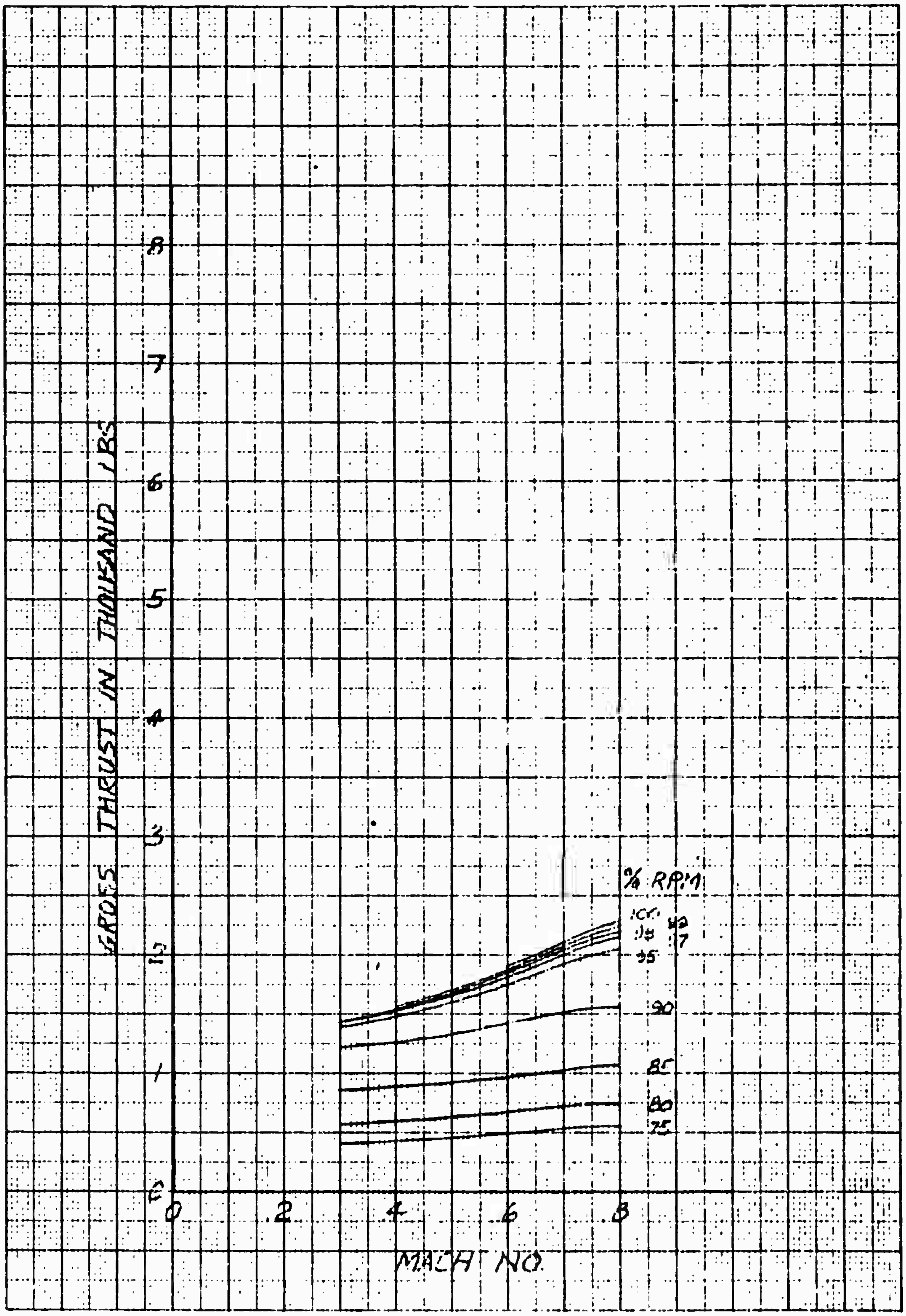

Figure 4.55 Gross Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Hot Day

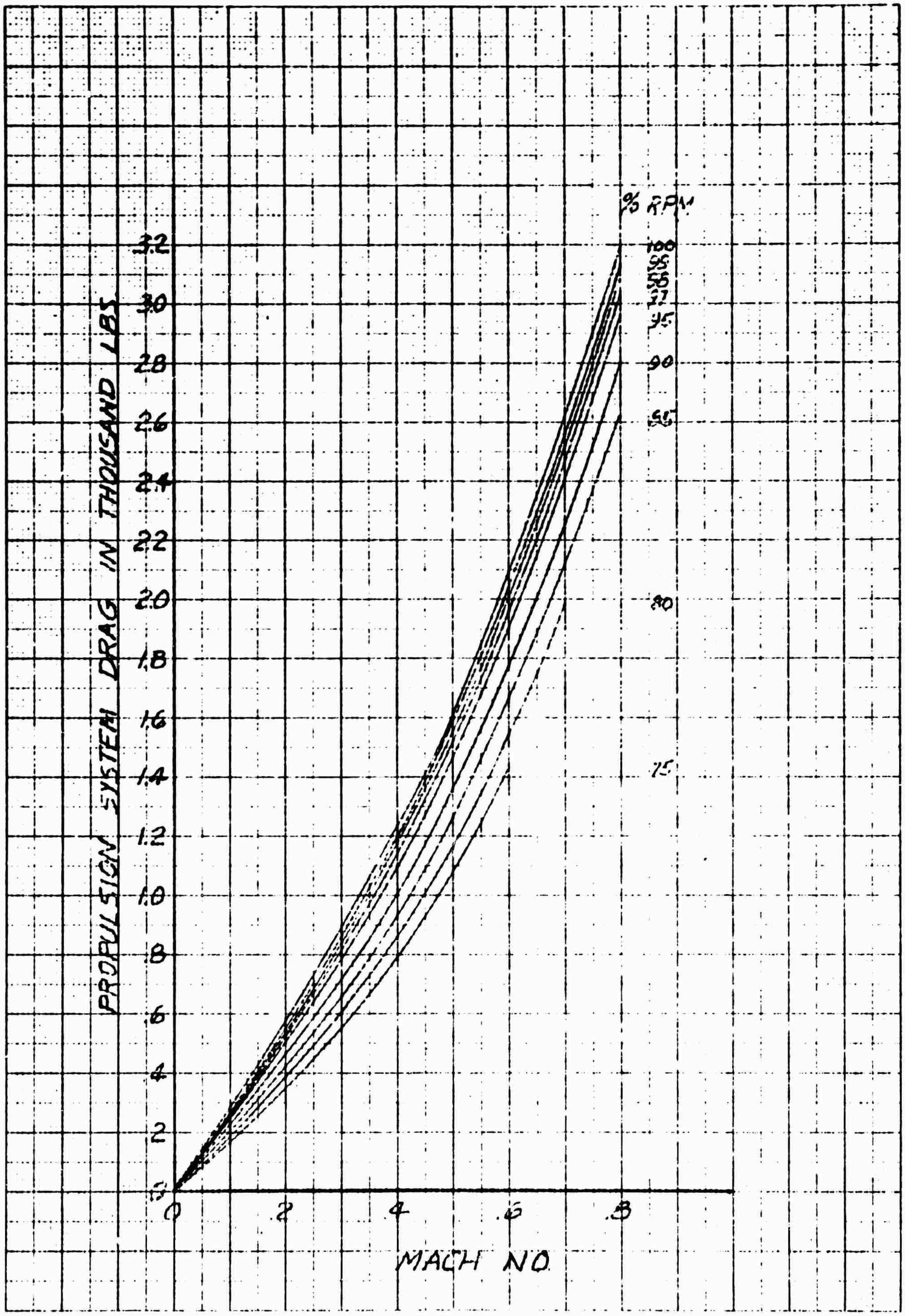

Figure 4.56 Propulsion System Drag vs Mach No. and % RPM;
Altitude = 0 ft., 2 Engines, Hot Day

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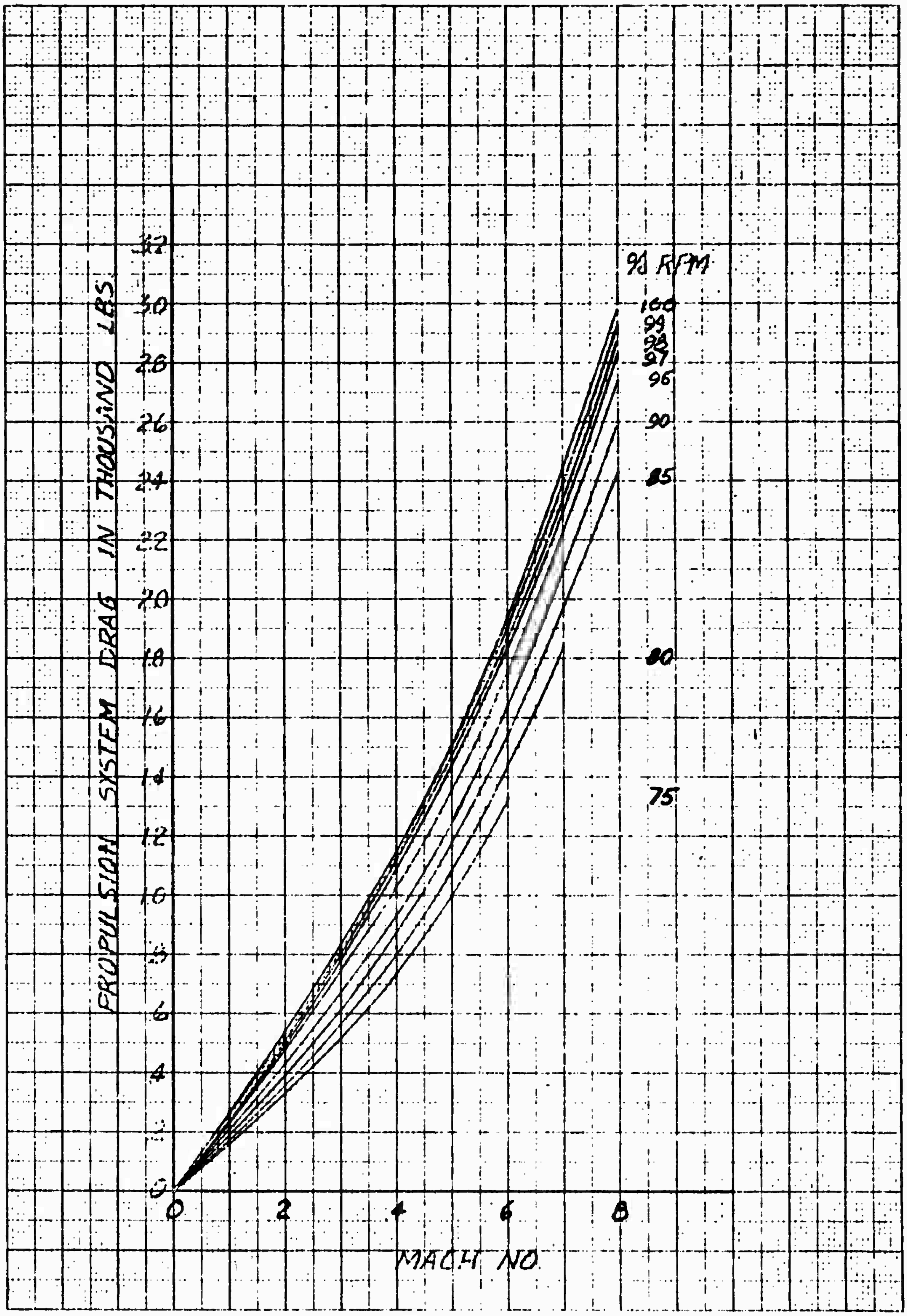

Figure 4.57 Propulsion System Drag vs Mach No. and % RPM; Altitude = 2500 ft., 2 Engines, Hot Day

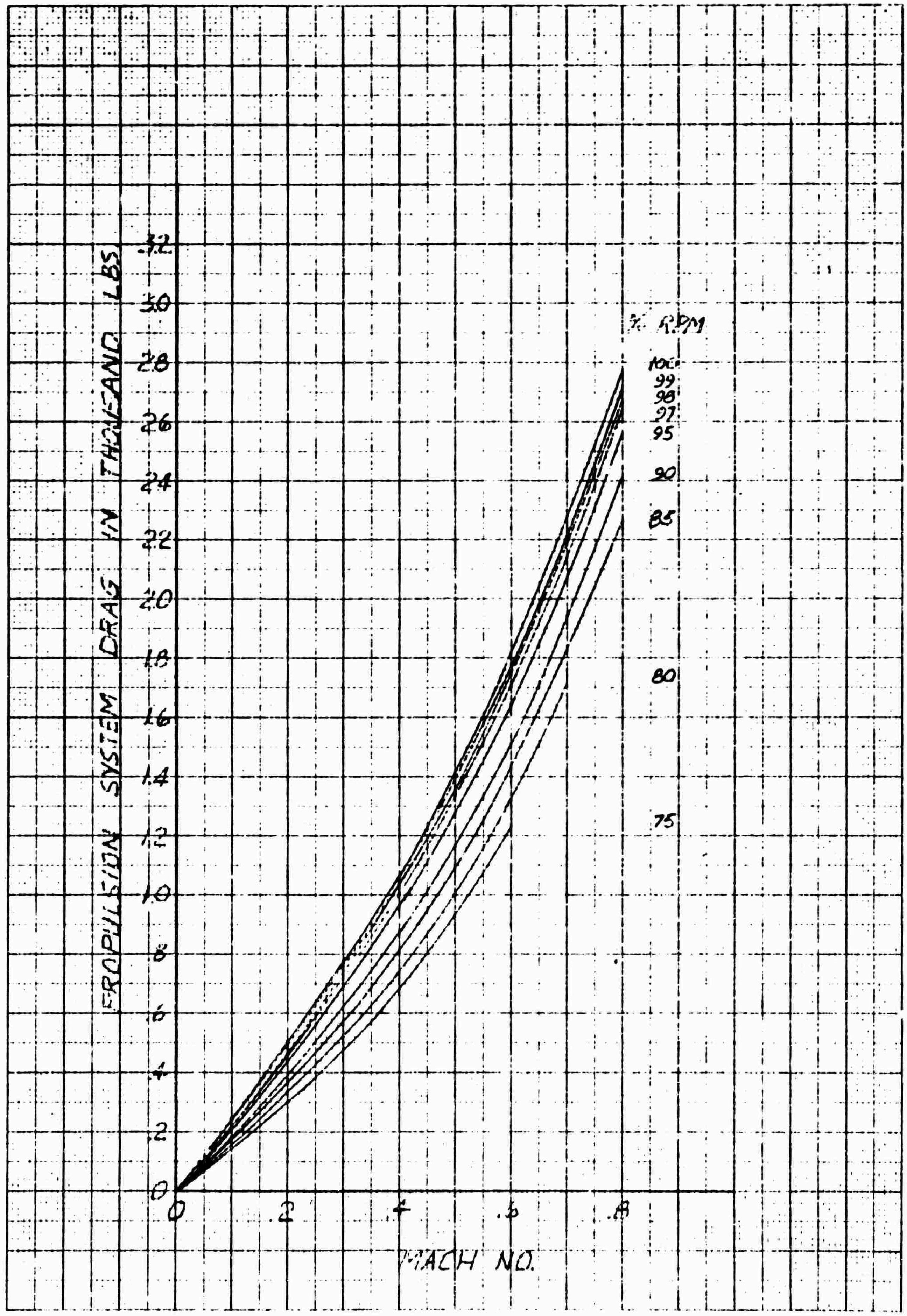

Figure 4.58 Propulsion System Drag vs Mach No. and % RPM; Altitude 5000 ft., 2 Engines, Hot Day

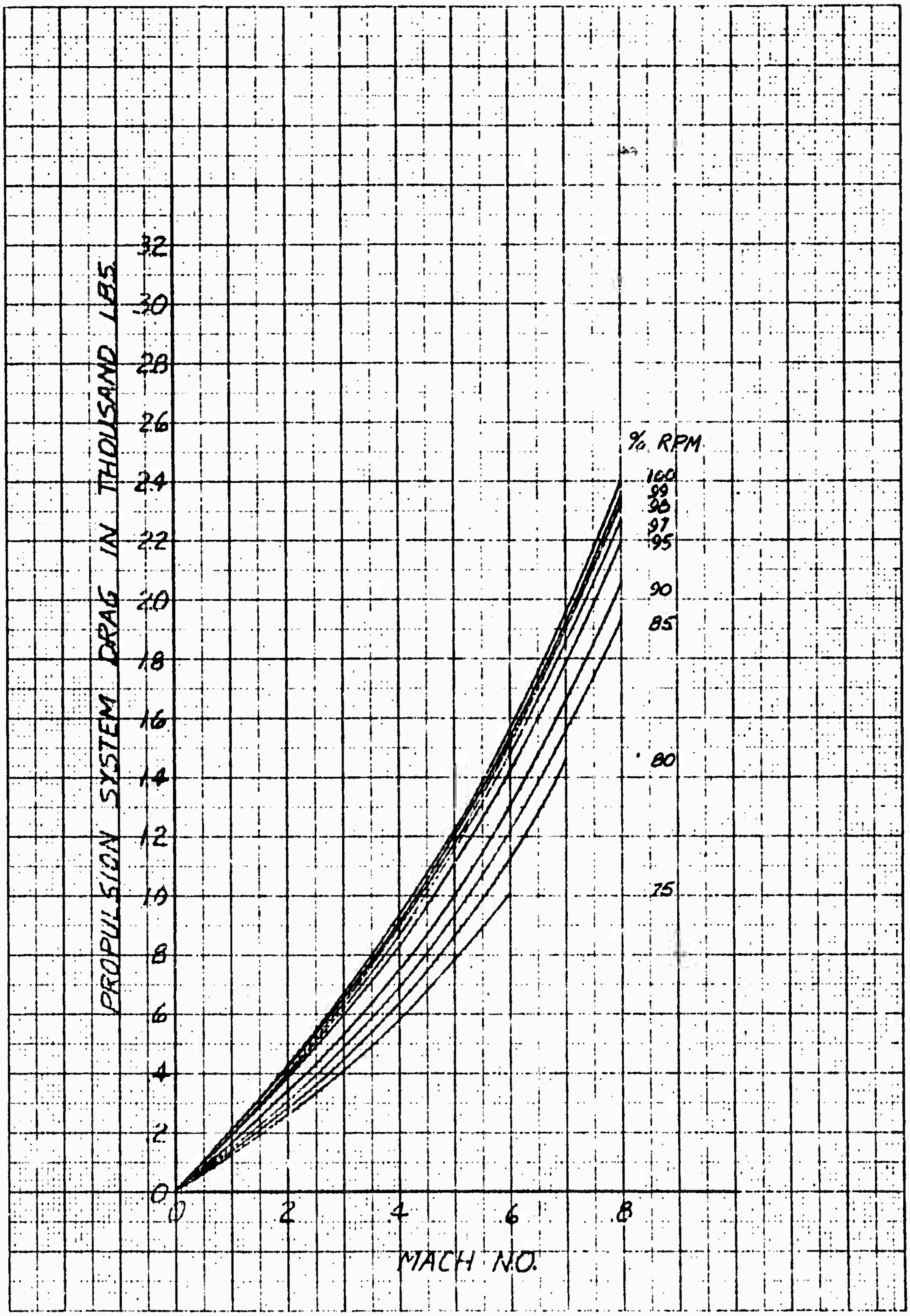

Figure 4.59 Propulsion System Drag vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Hot Day

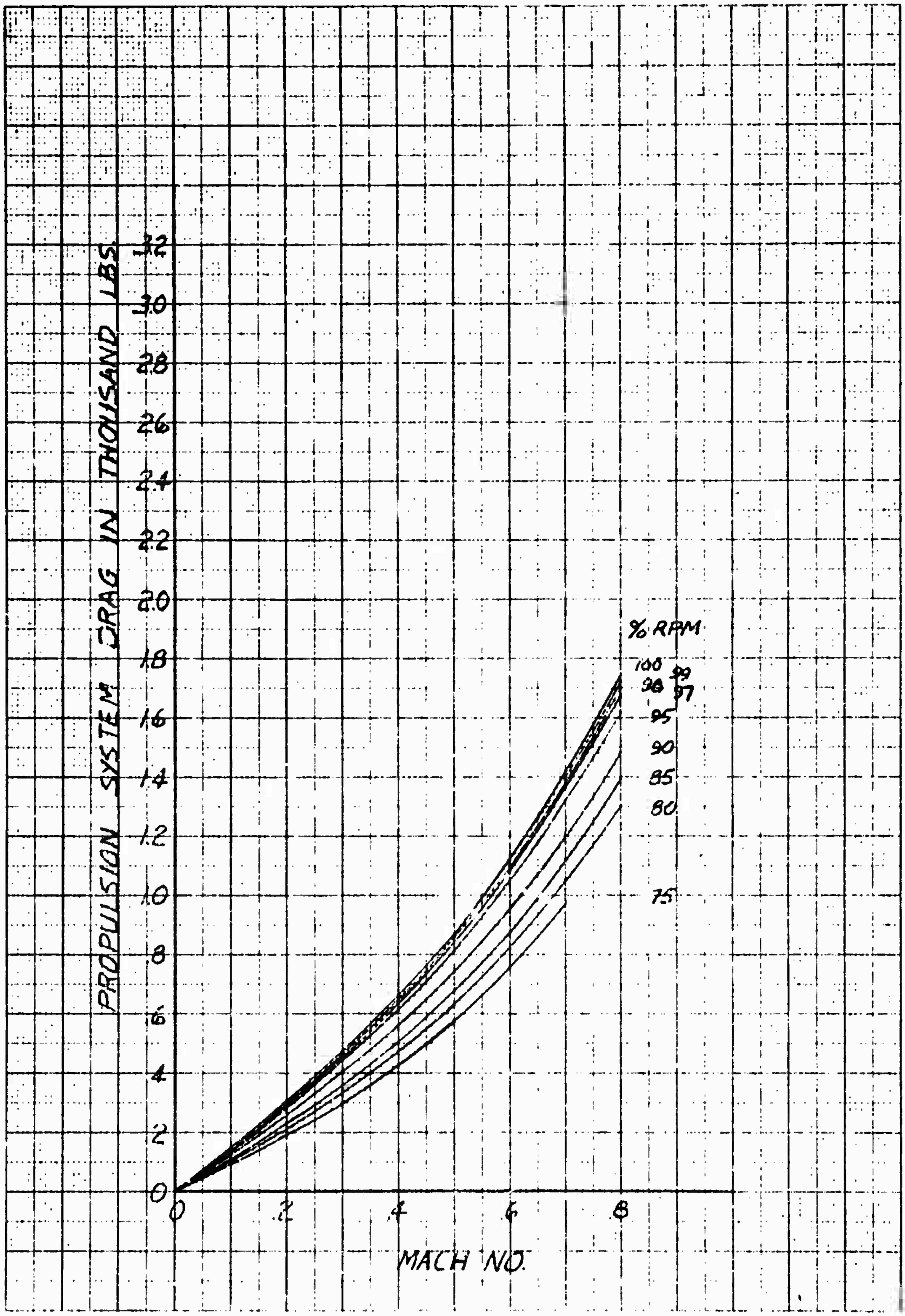

Figure 4.60 Propulsion System Drag vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Hot Day

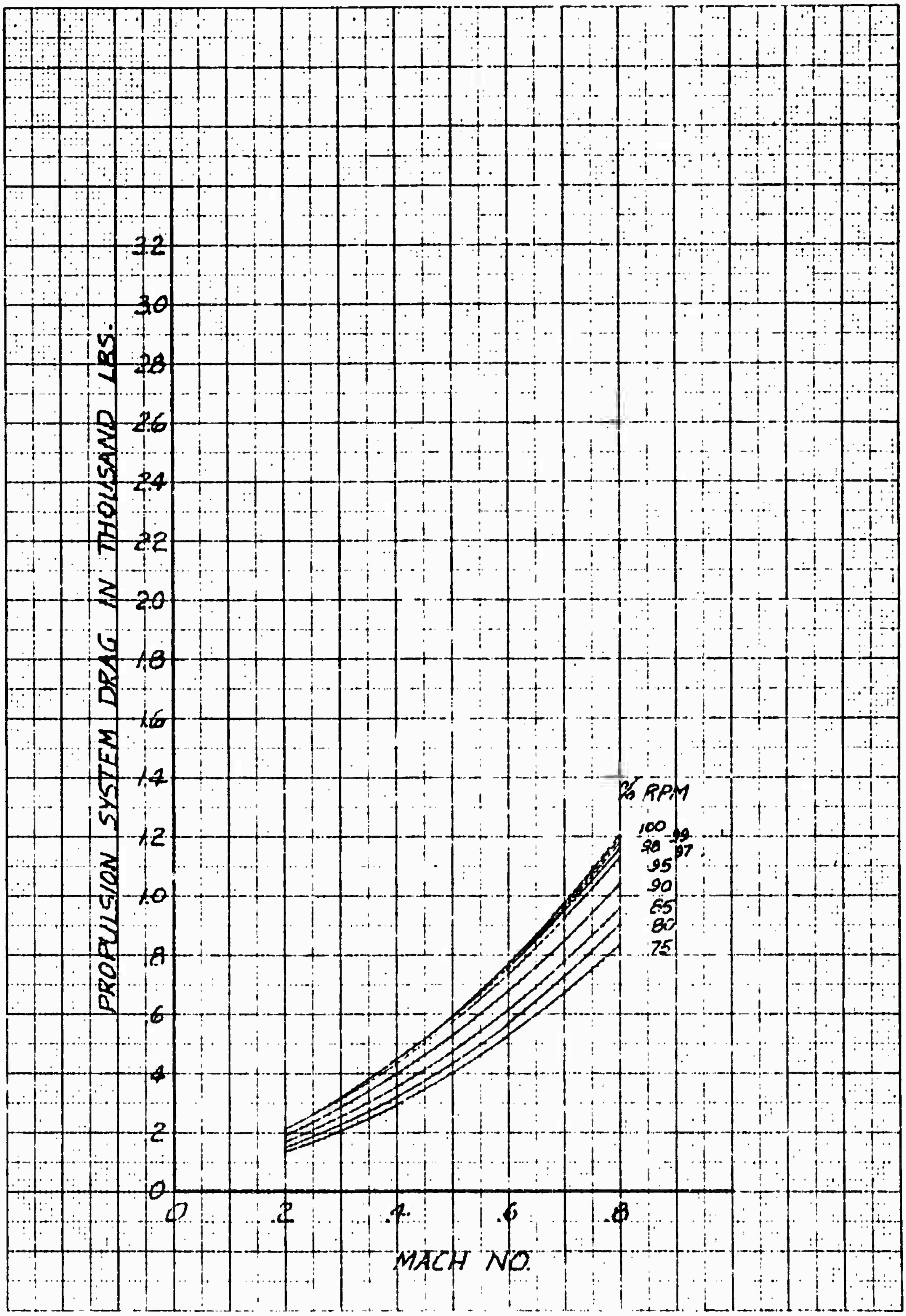

Figure 4.61 Propulsion System Drag vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Hot Day

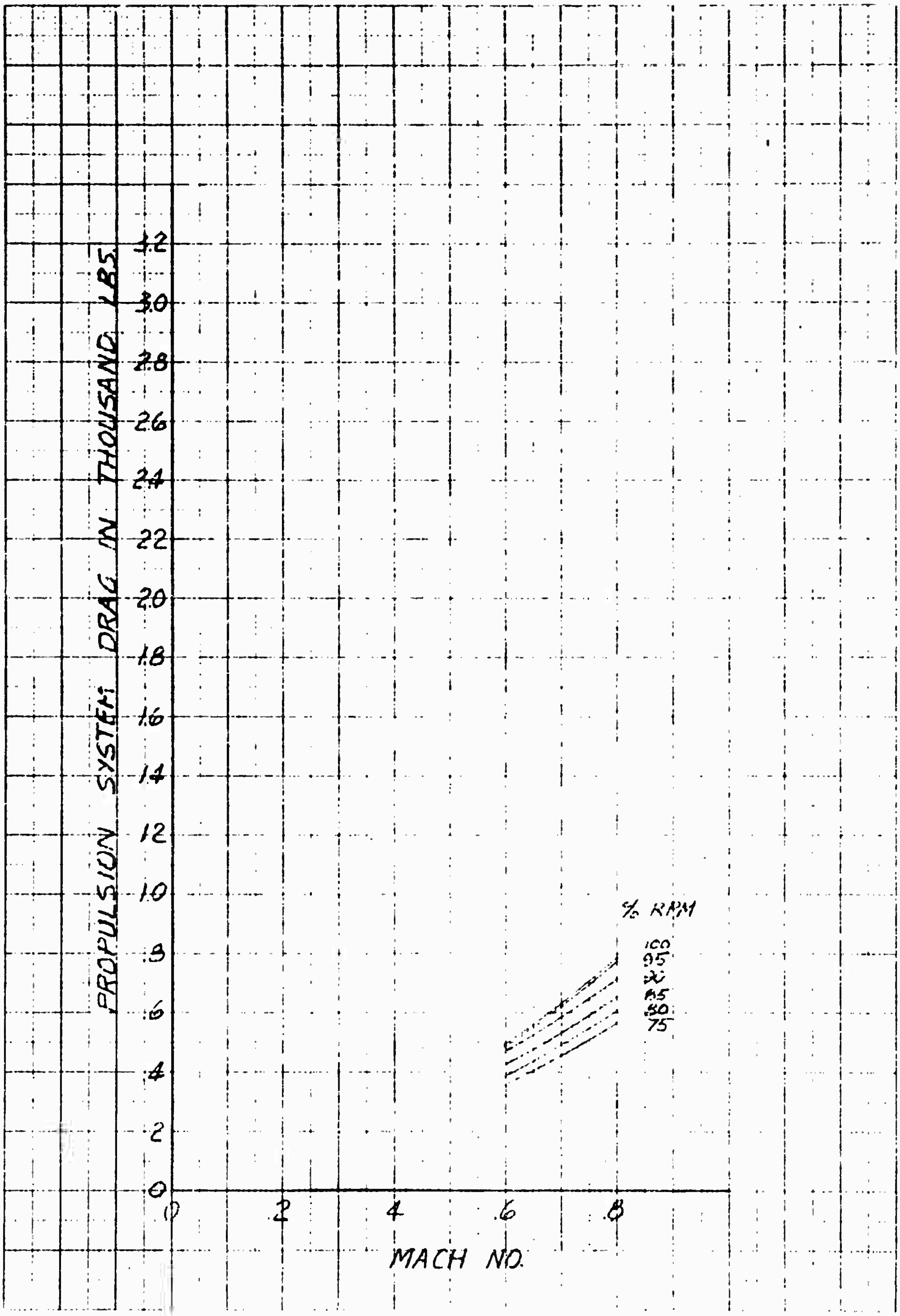

Figure 4.62 Propulsion System Drag vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Hot Day

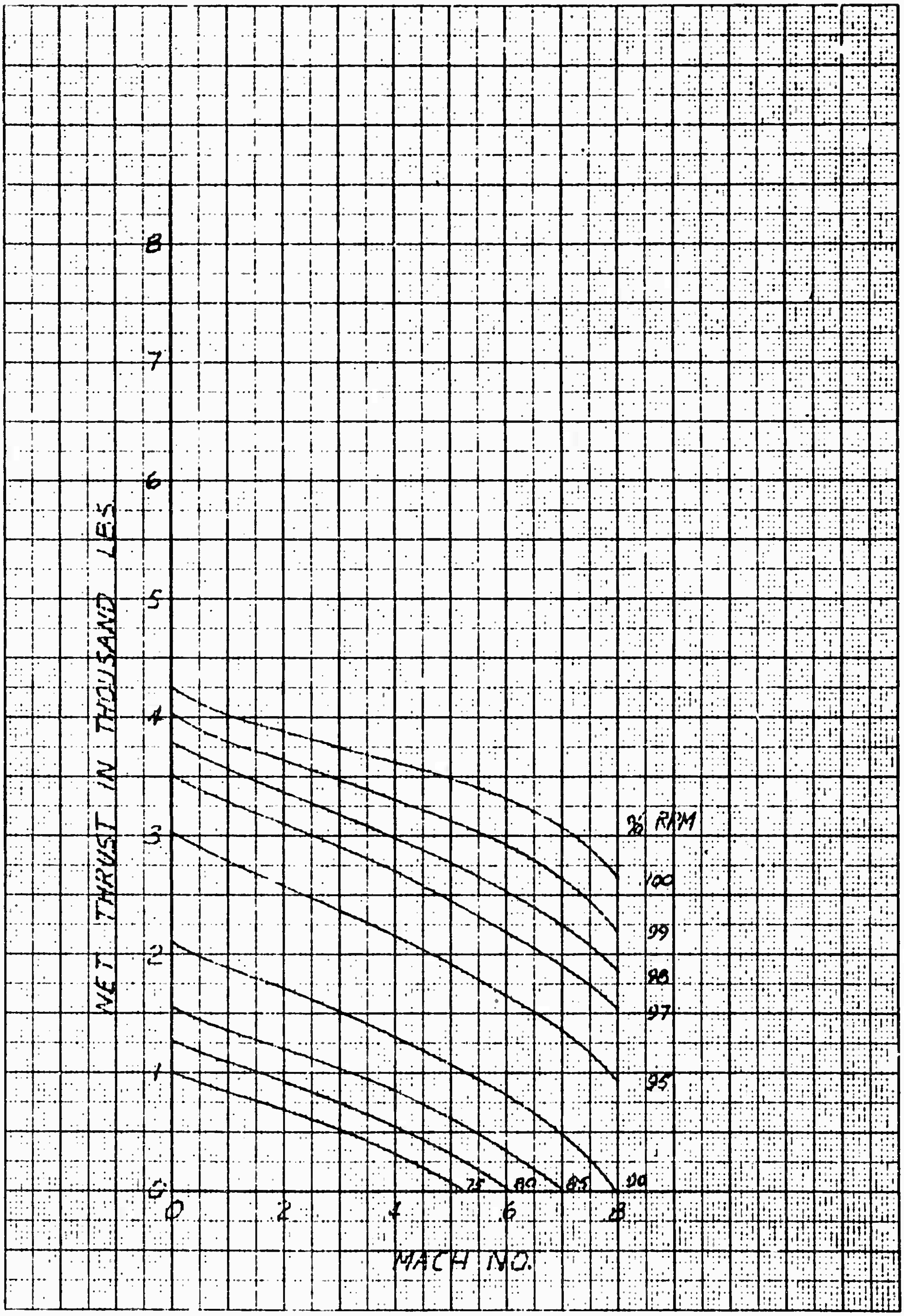

Figure 4.63 Net Thrust vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Hot Day

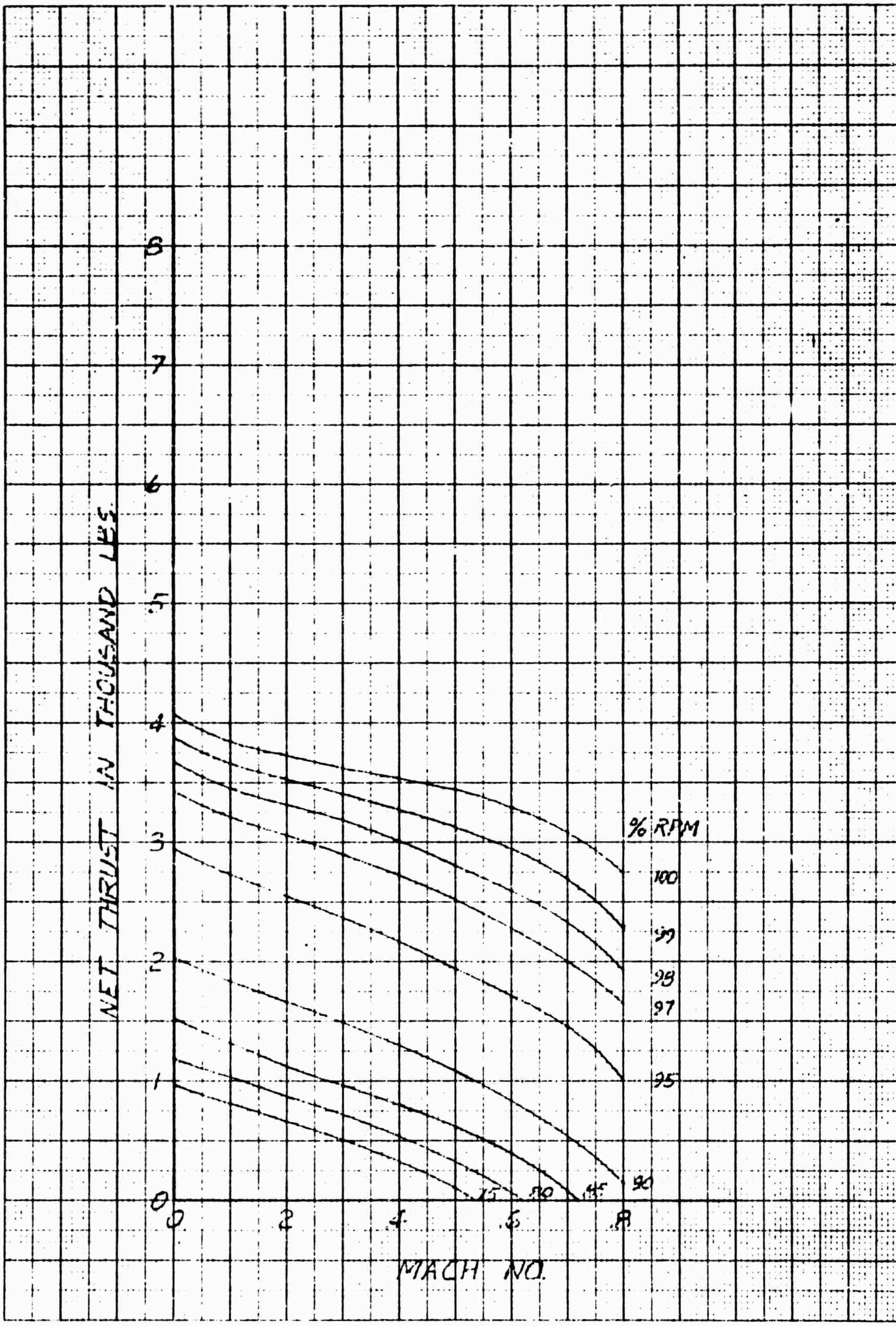

Figure 4.64 Net Thrust vs Mach No. and % RPM; Altitude = 2500 ft., 2 Engines, Hot Day

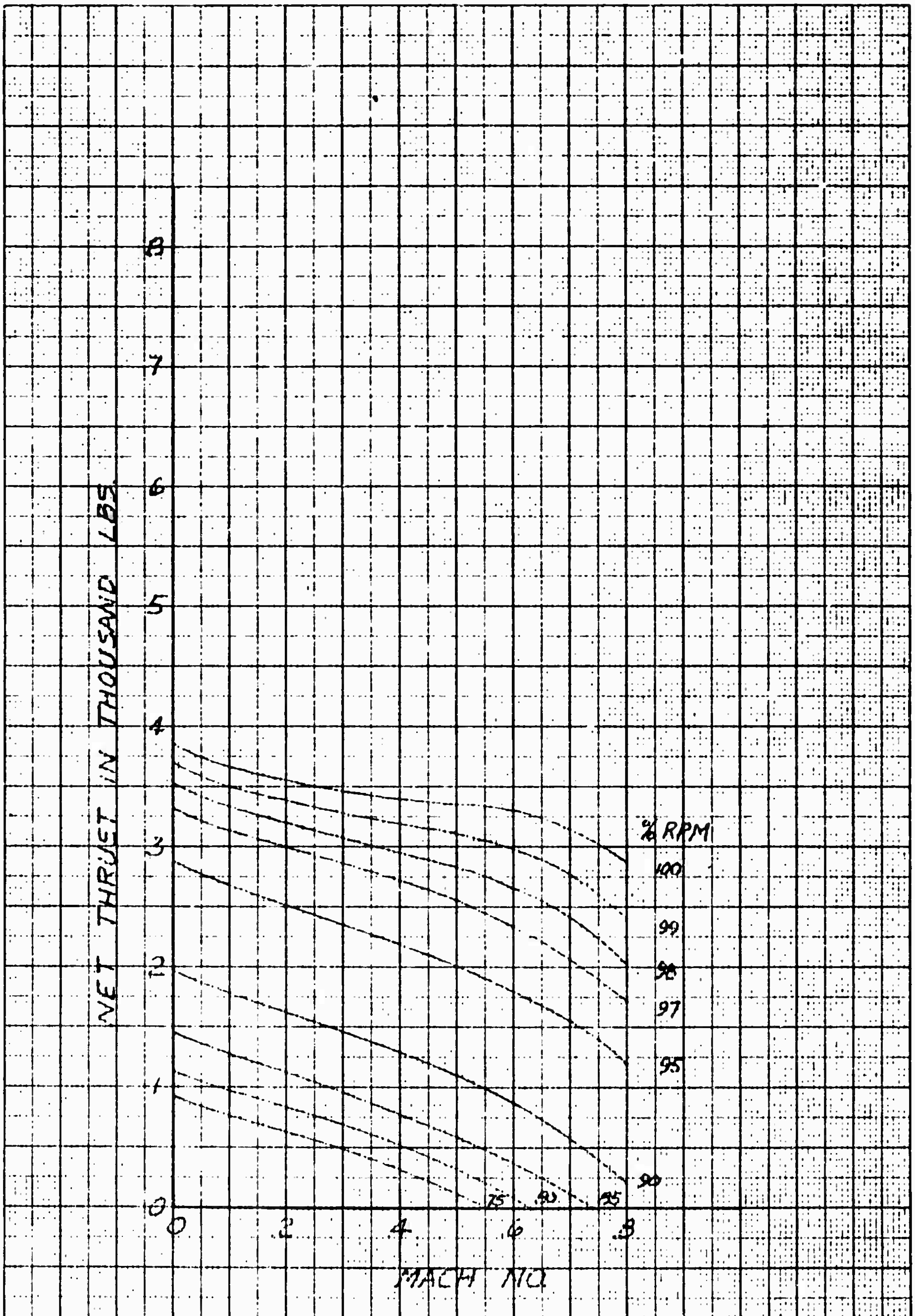

Figure 4.65 Net Thrust vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Hot Day

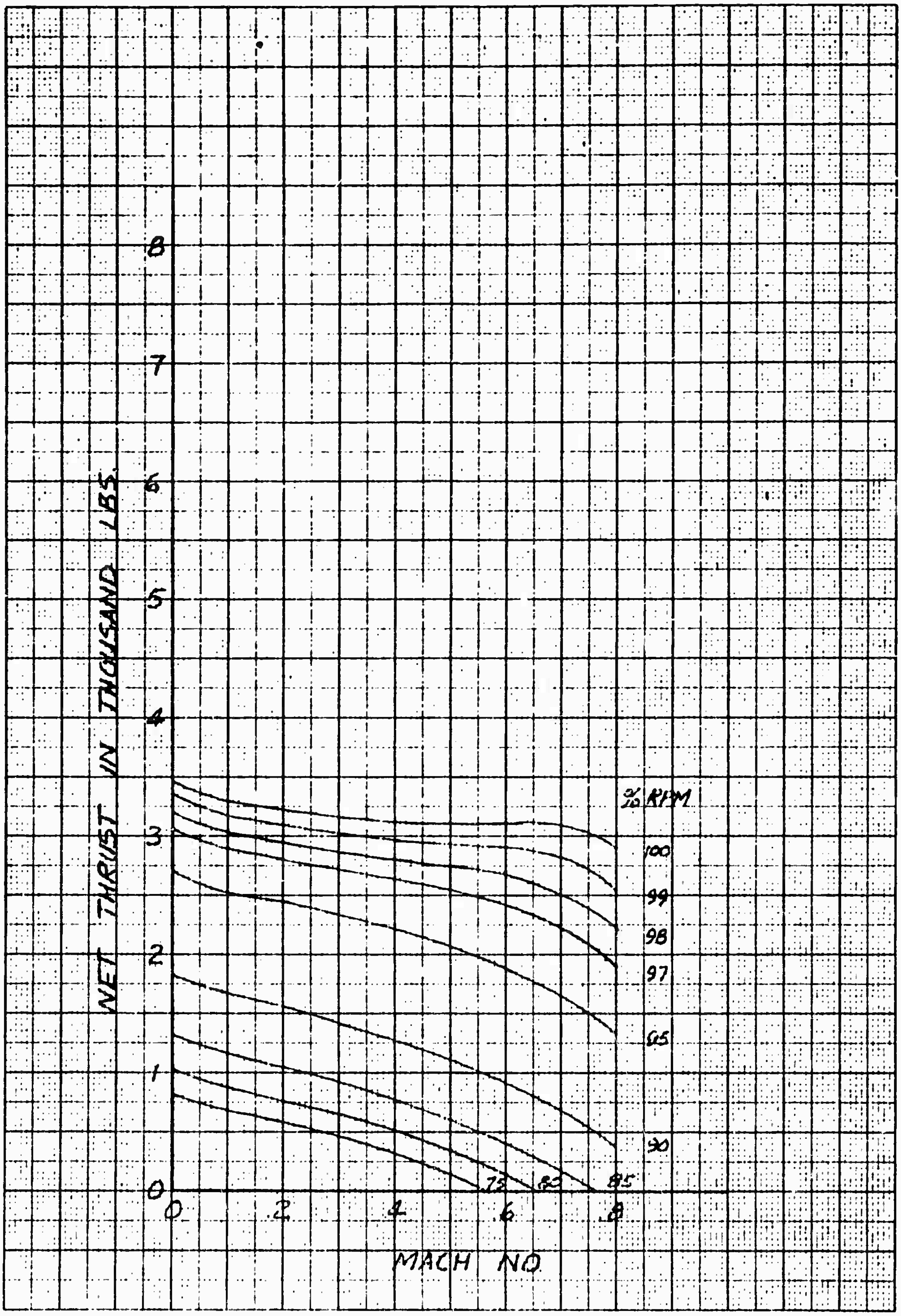

Figure 4.66 Net Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Hot Day

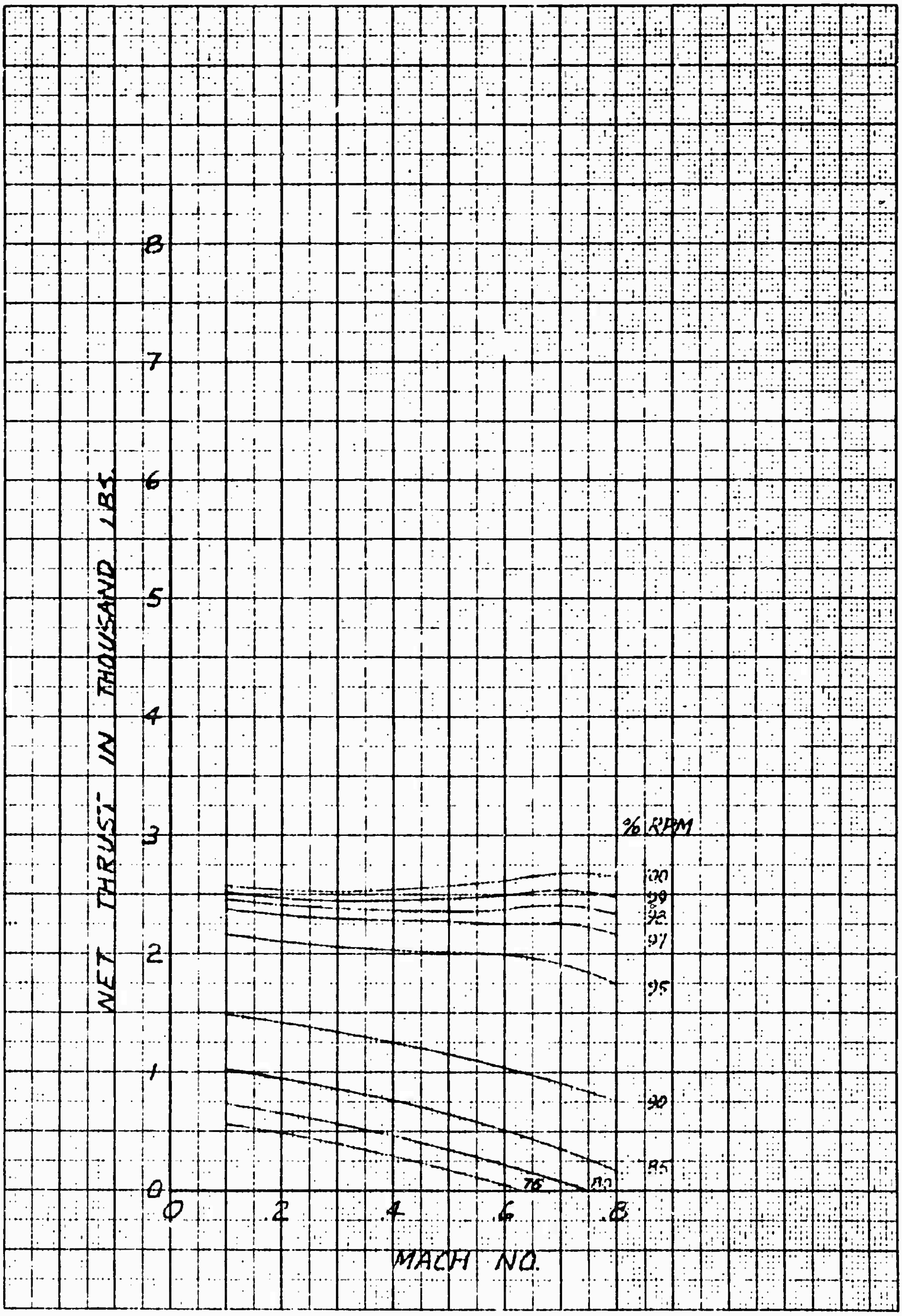

Figure 4.67 Net Thrust vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Hot Day

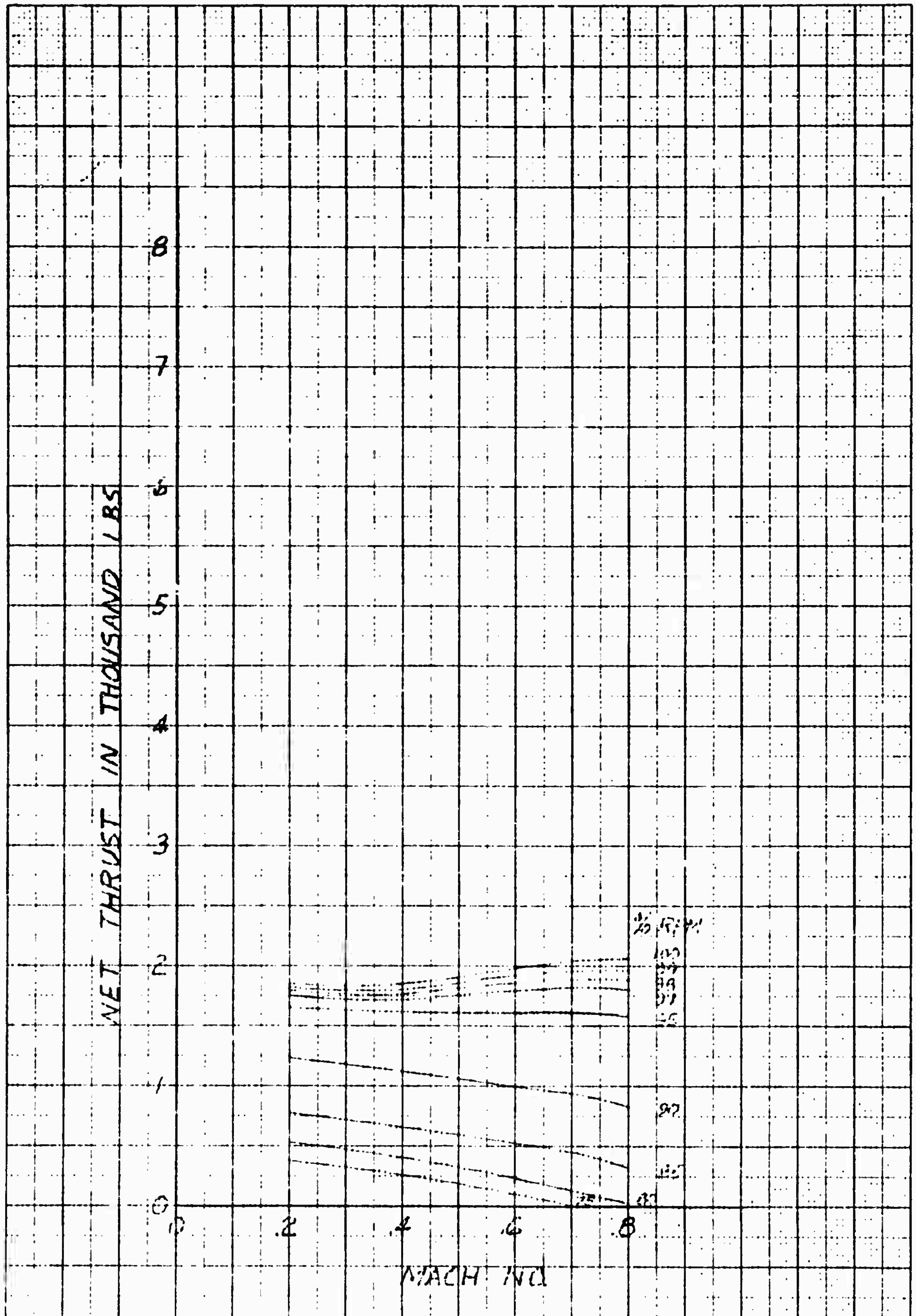

Figure 4.68 Net Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Hot Day

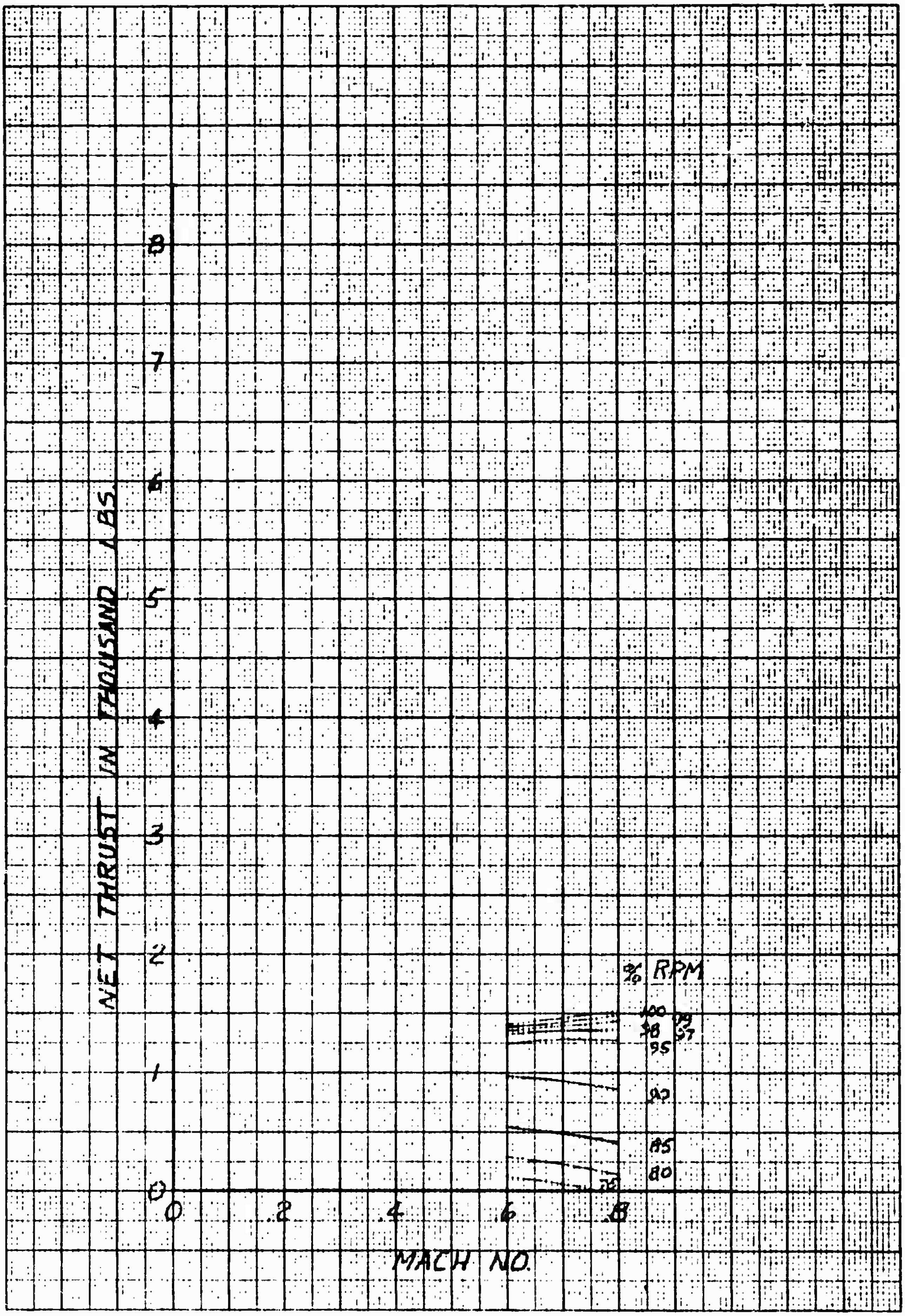

Figure 4.69 Net Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Hot Day

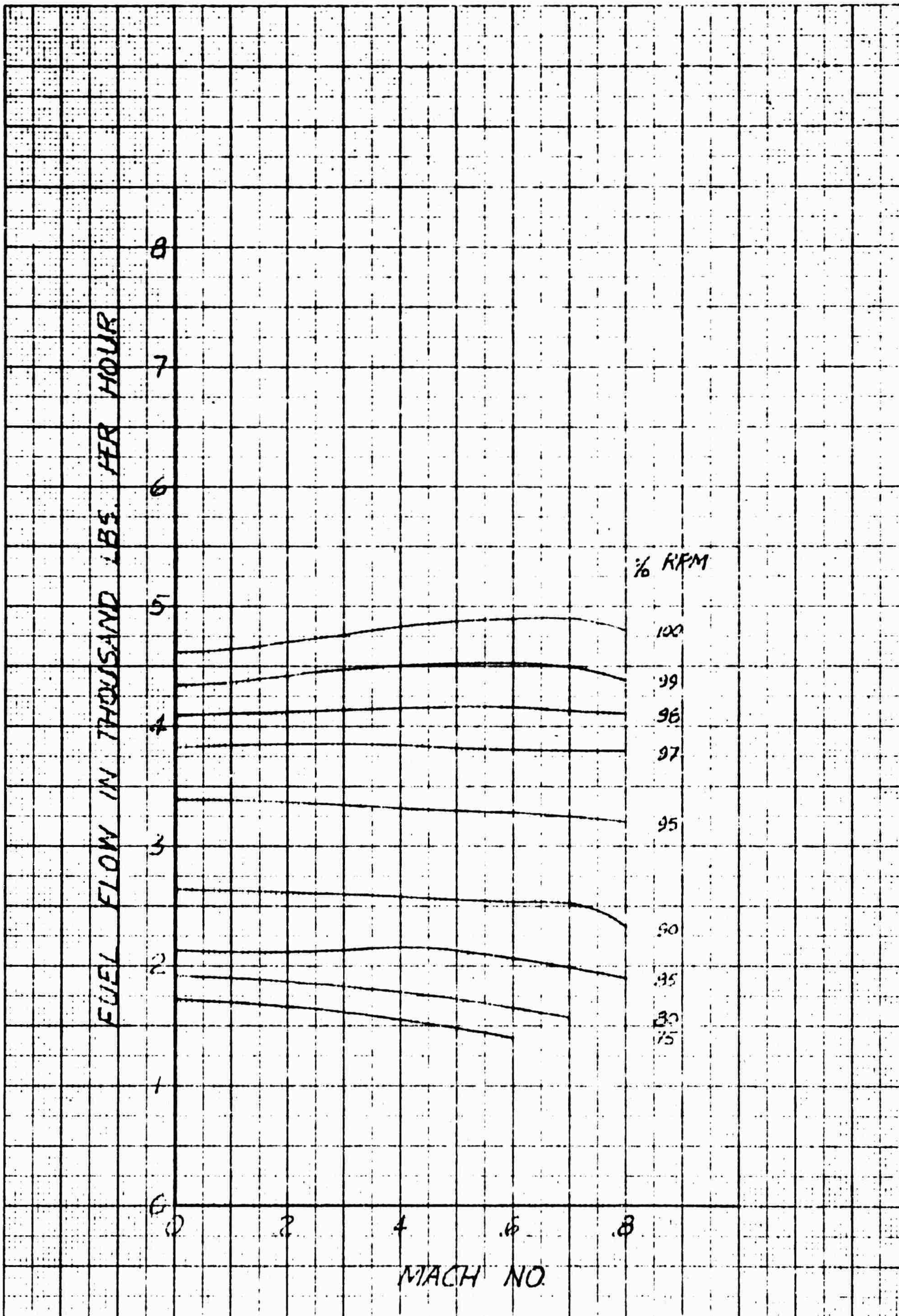

Figure 4.70 Fuel Flow vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Hot Day

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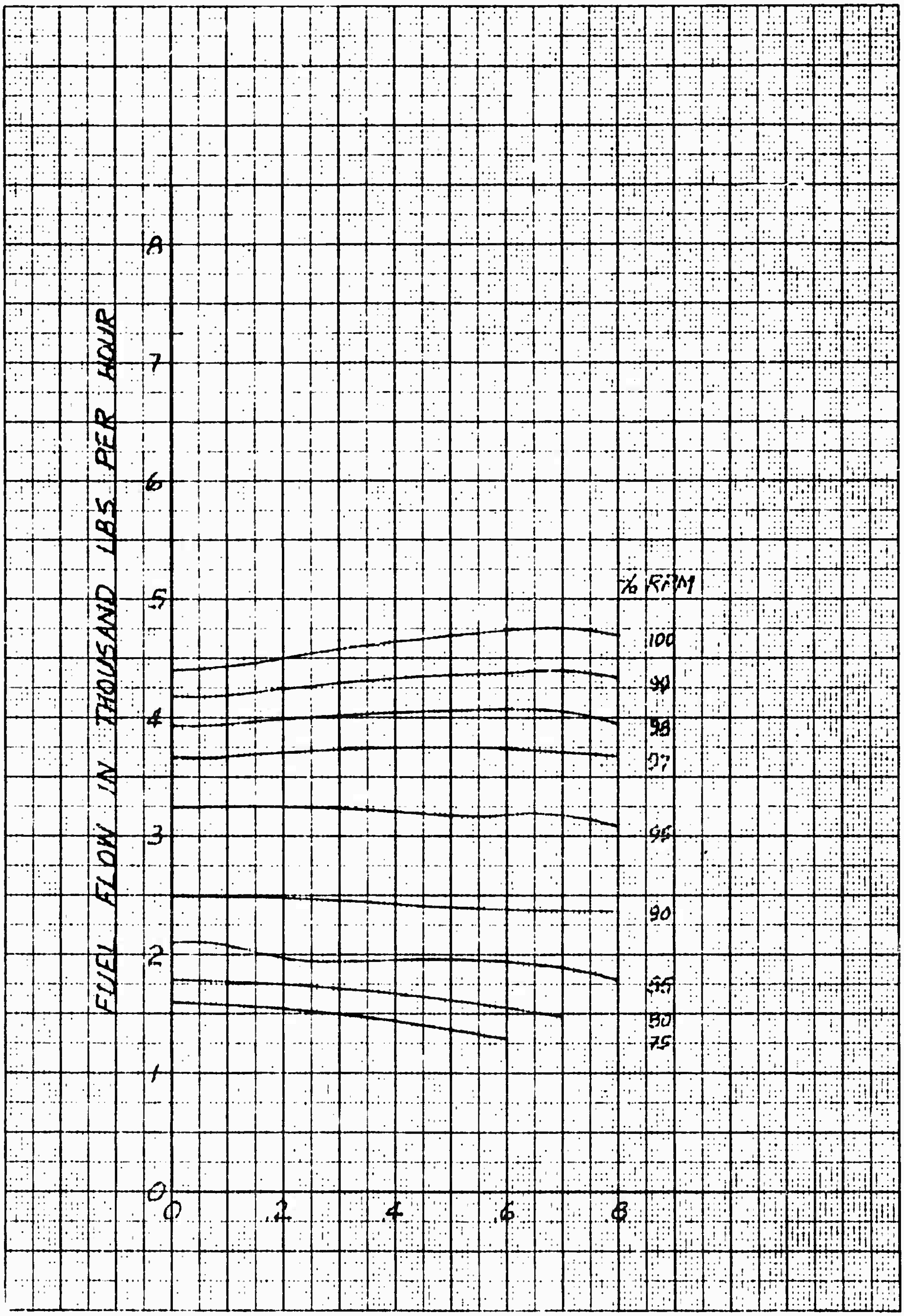

Figure 4.71 Fuel Flow vs Mach No. and % RPM; Altitude = 2500 ft., 2 Engines, Hot Day

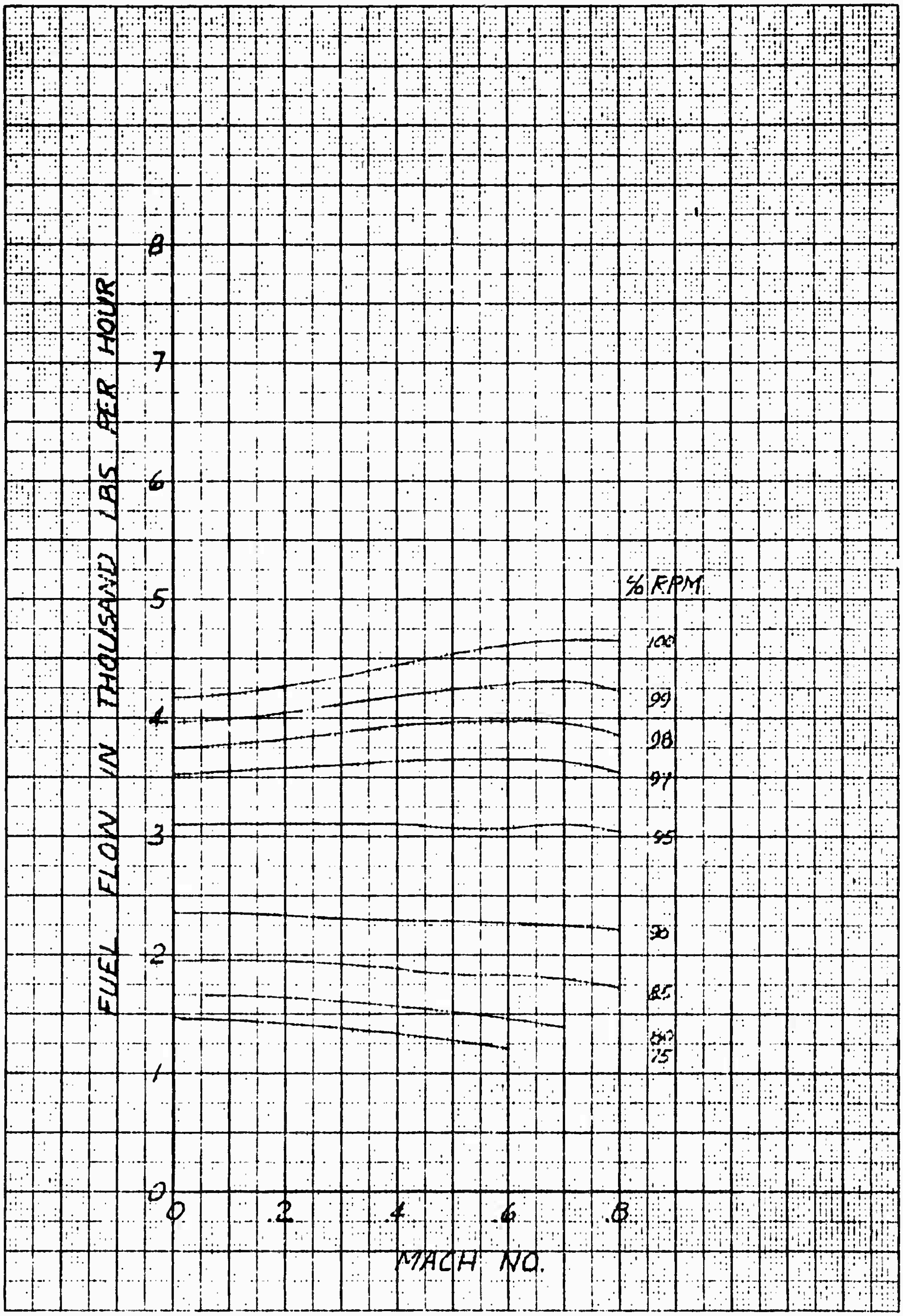

Figure 4.72 Fuel Flow vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Hot Day

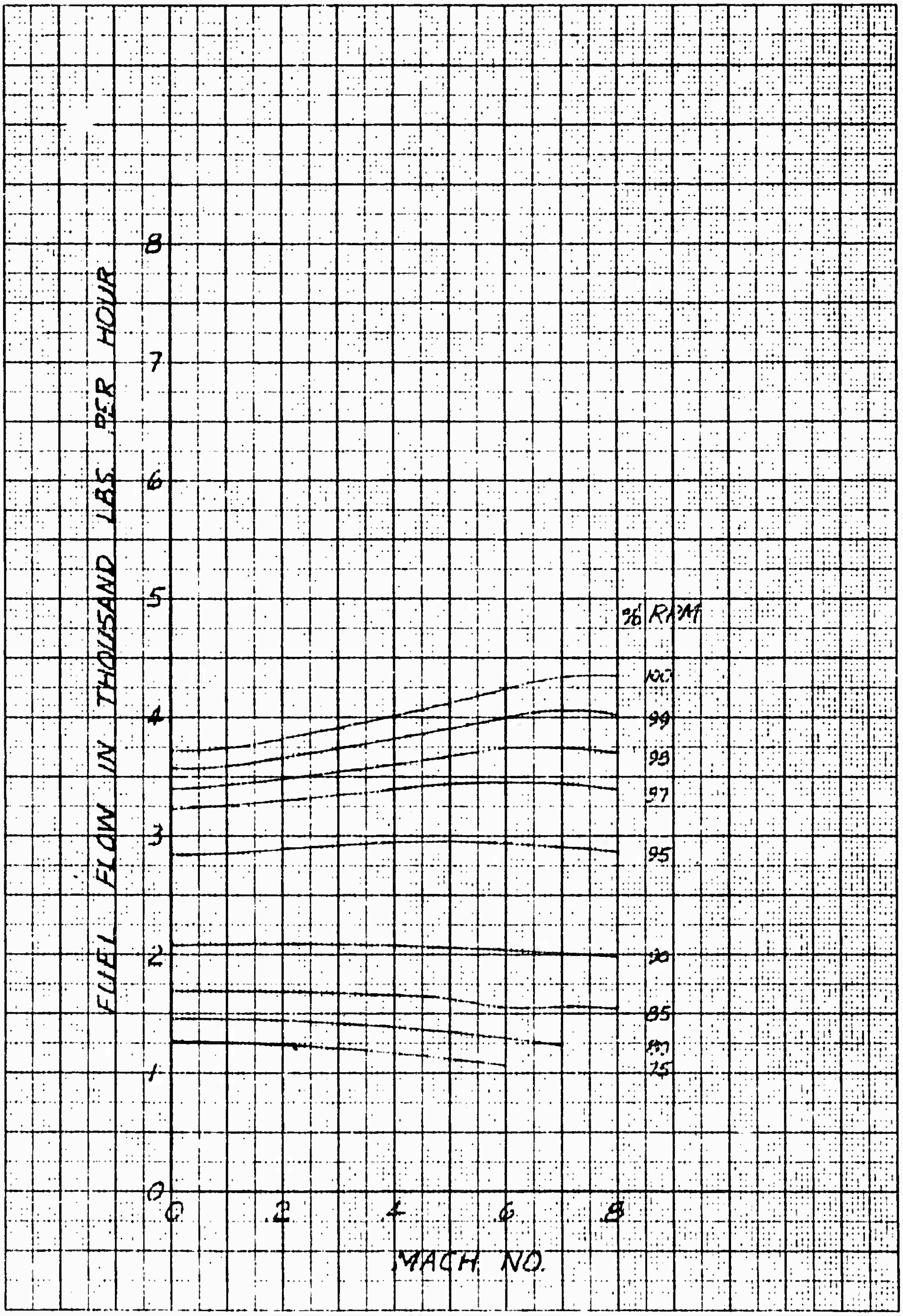

Figure 4.73 Fuel Flow vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Hot Day

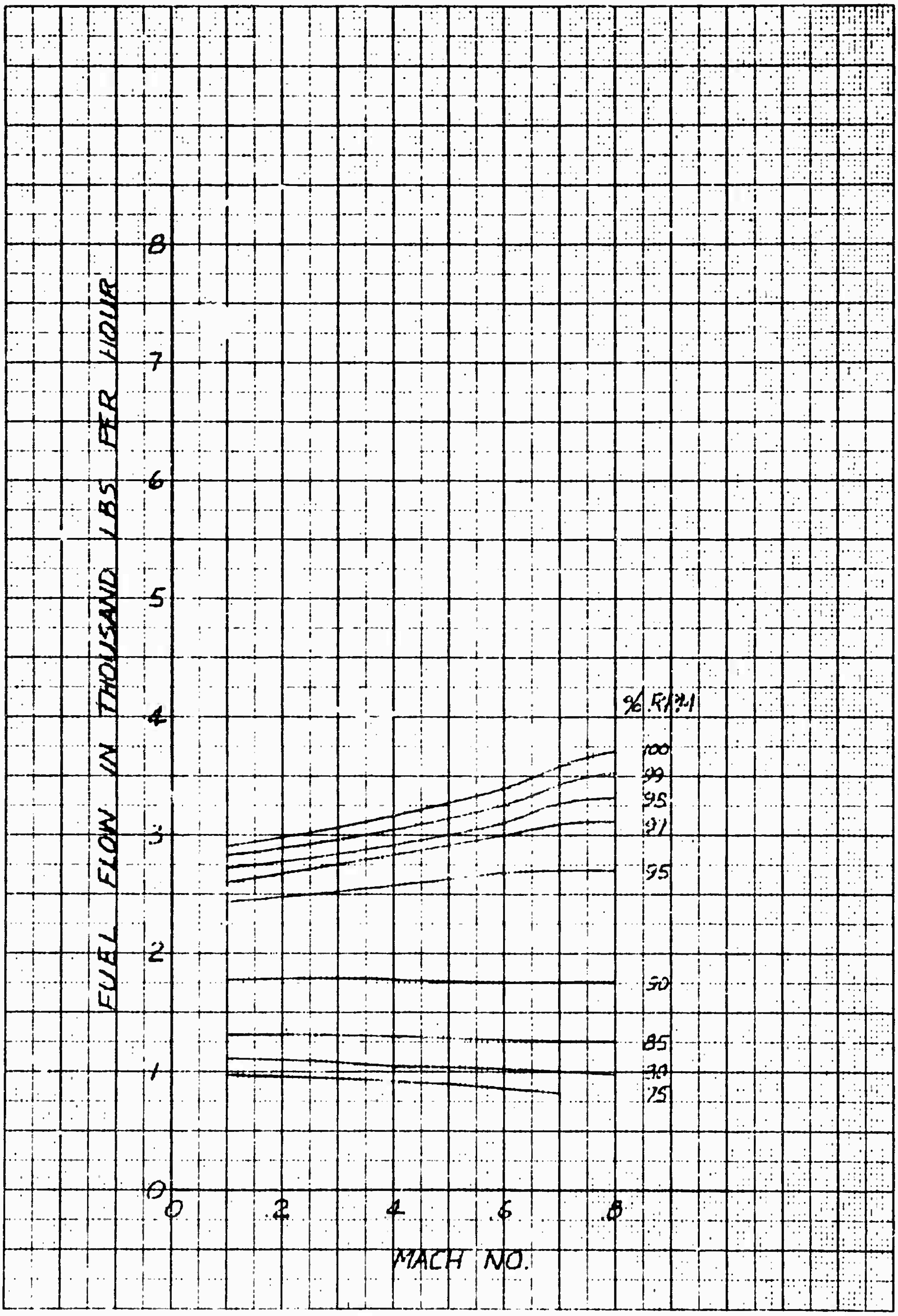

Figure 4.74 Fuel Flow vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Hot Day

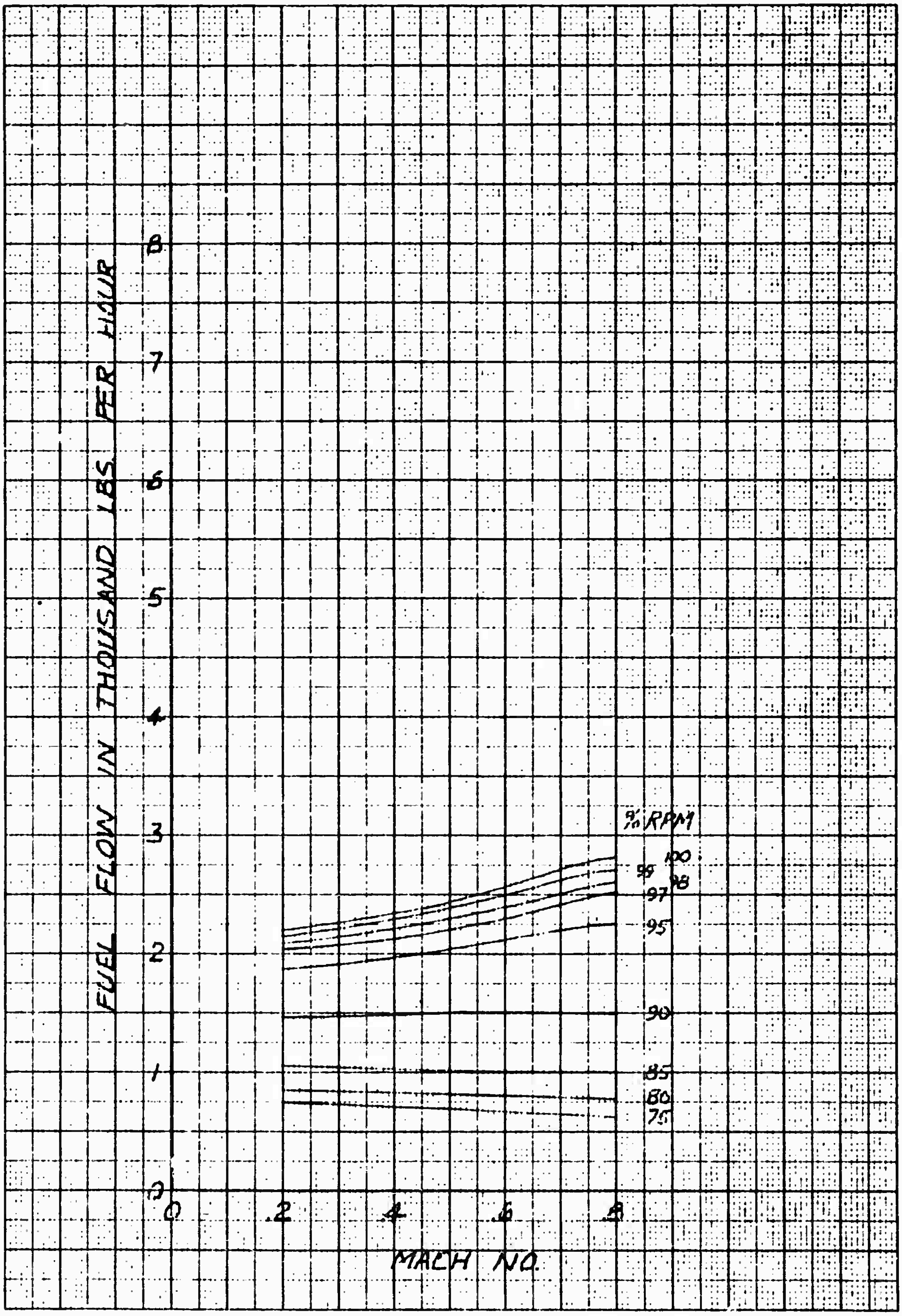

Figure 4.75 Fuel Flow vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Hot Day

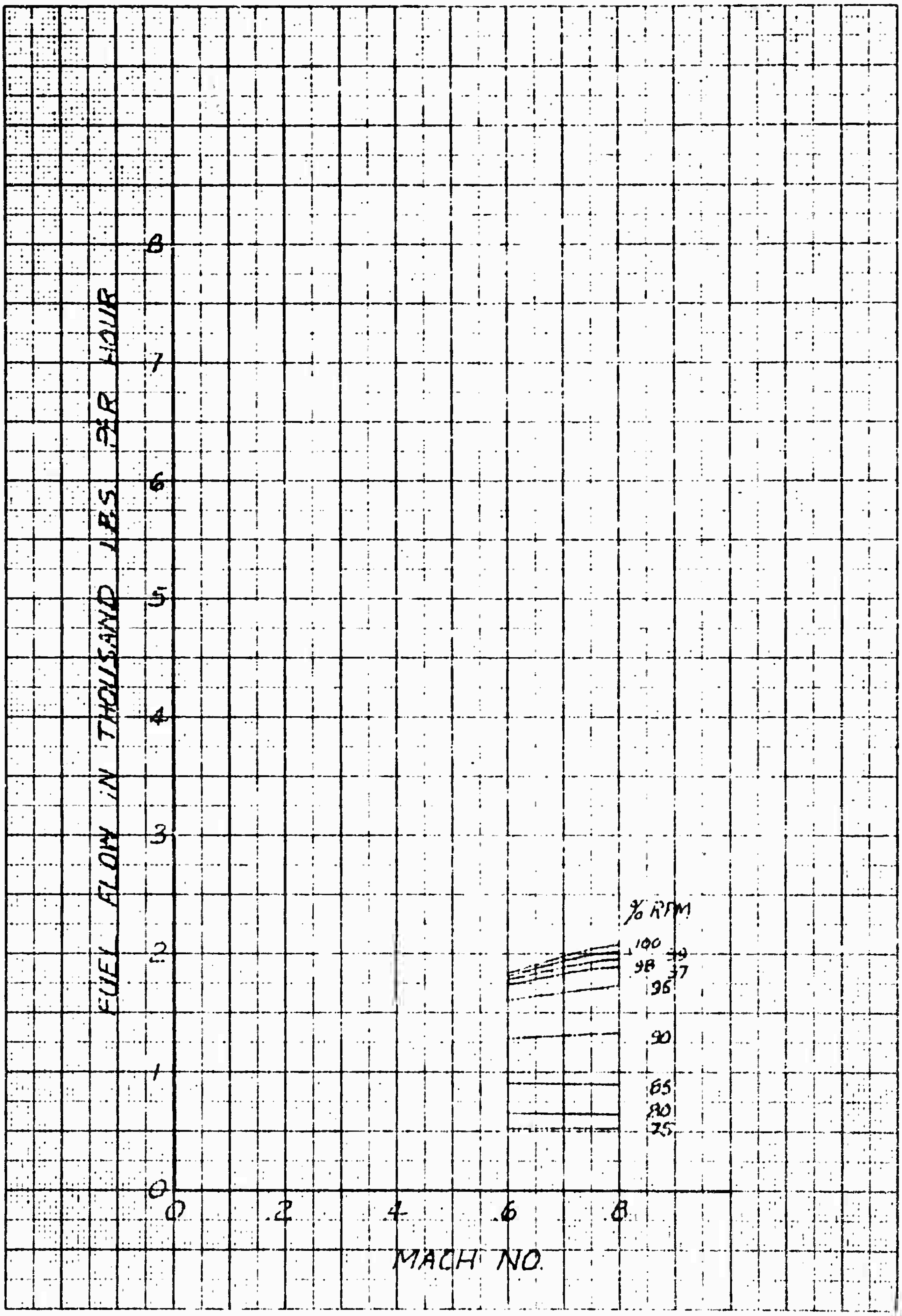

Figure 4.76 Fuel Flow vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Hot Day

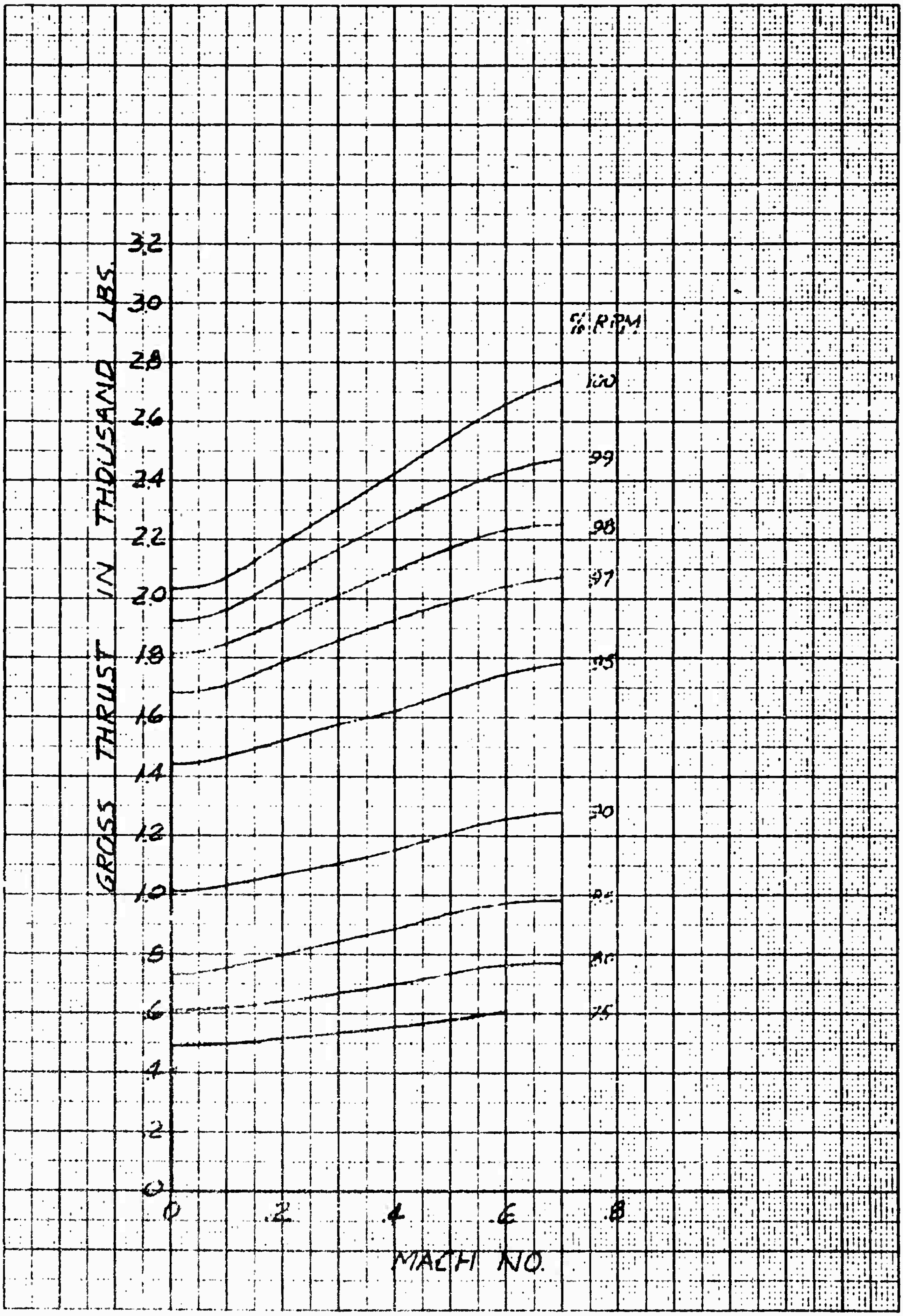

Figure 4.77 Gross Thrust vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Hot Day

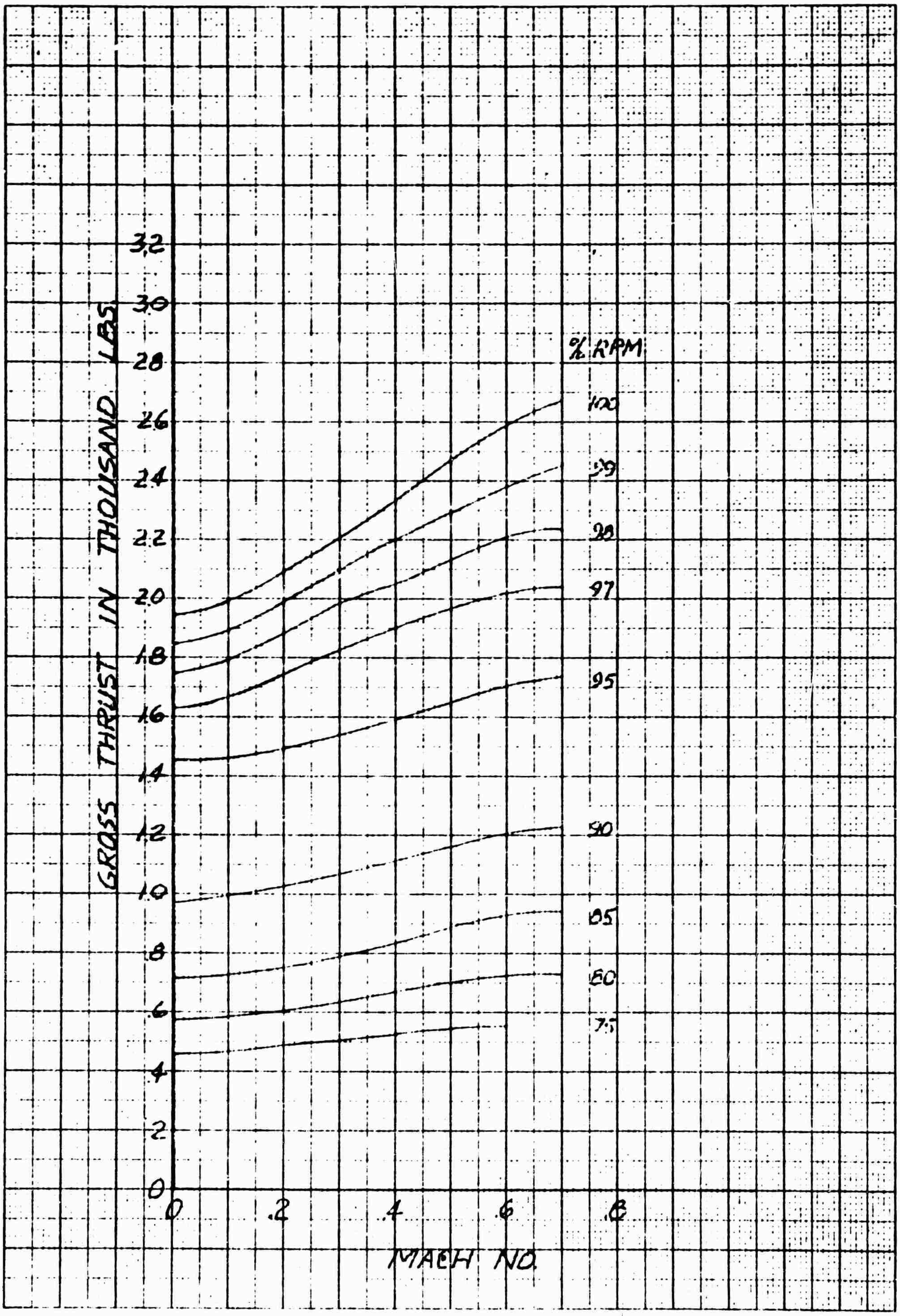

Figure 4.78 Gross Thrust vs Mach No. and % RPM; Altitude = 2500 ft., 1 Engine, Hot Day

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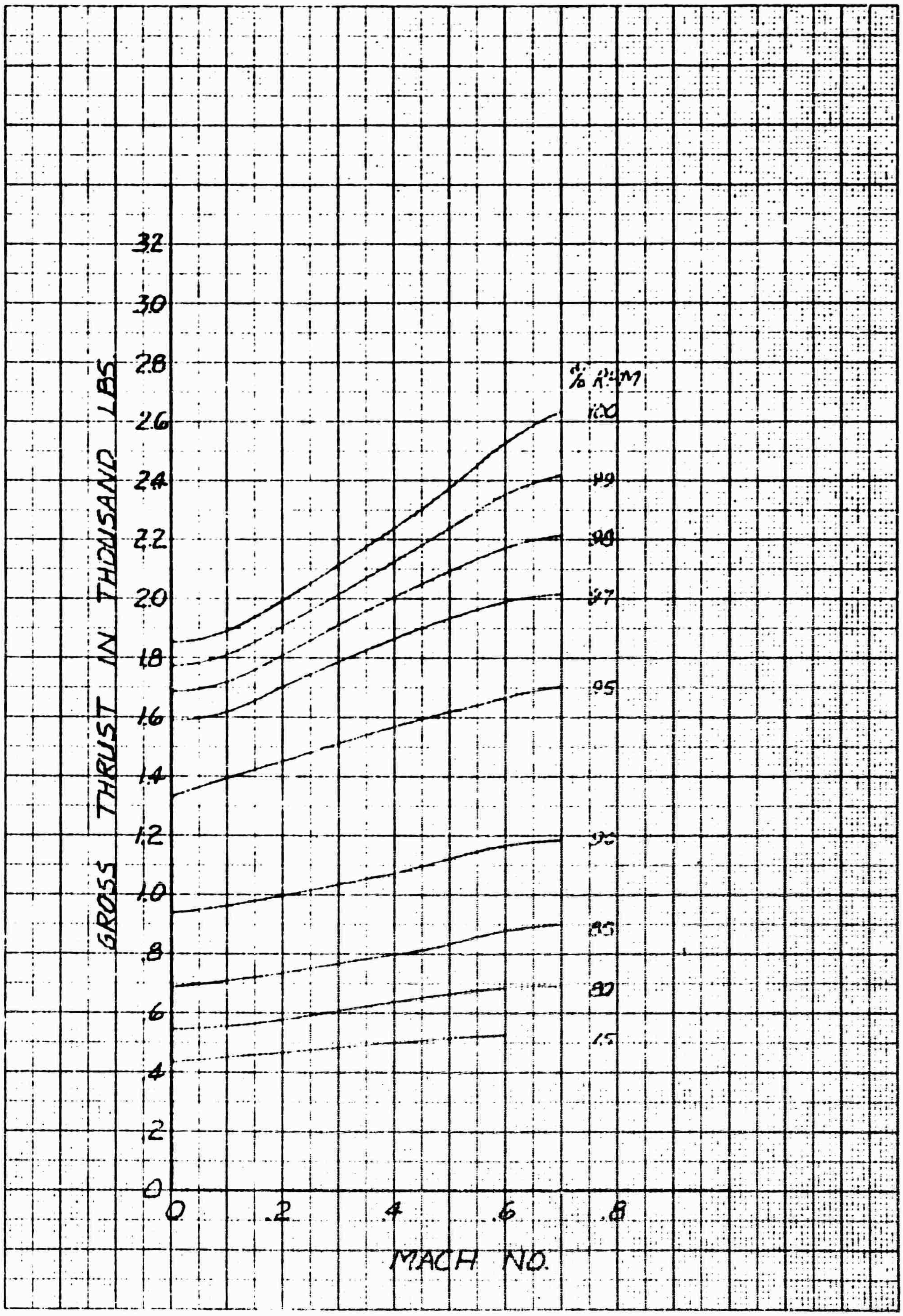

Figure 4.79 Gross Thrust vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Hot Day

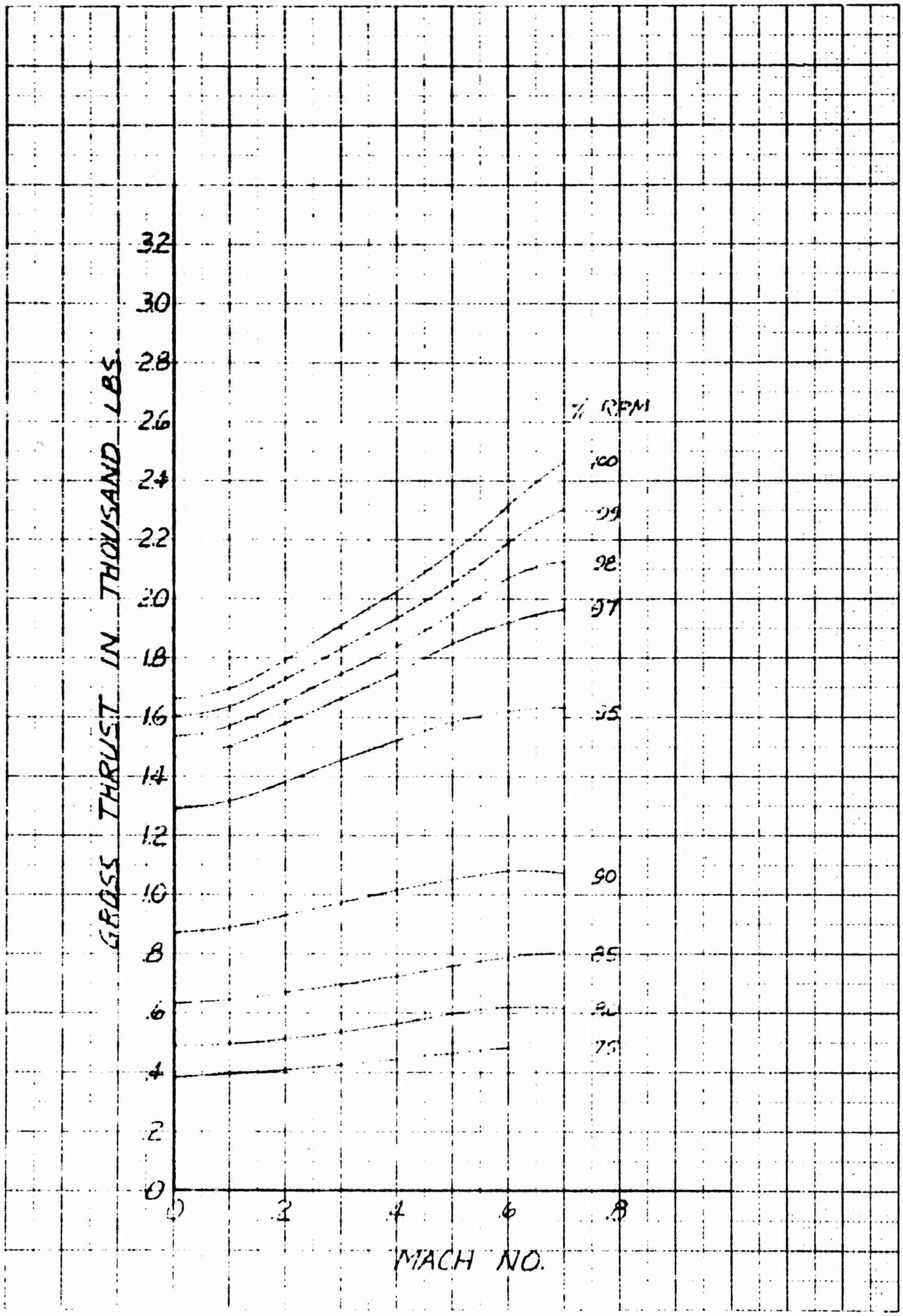

Figure 4.80 Gross Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Hot Day

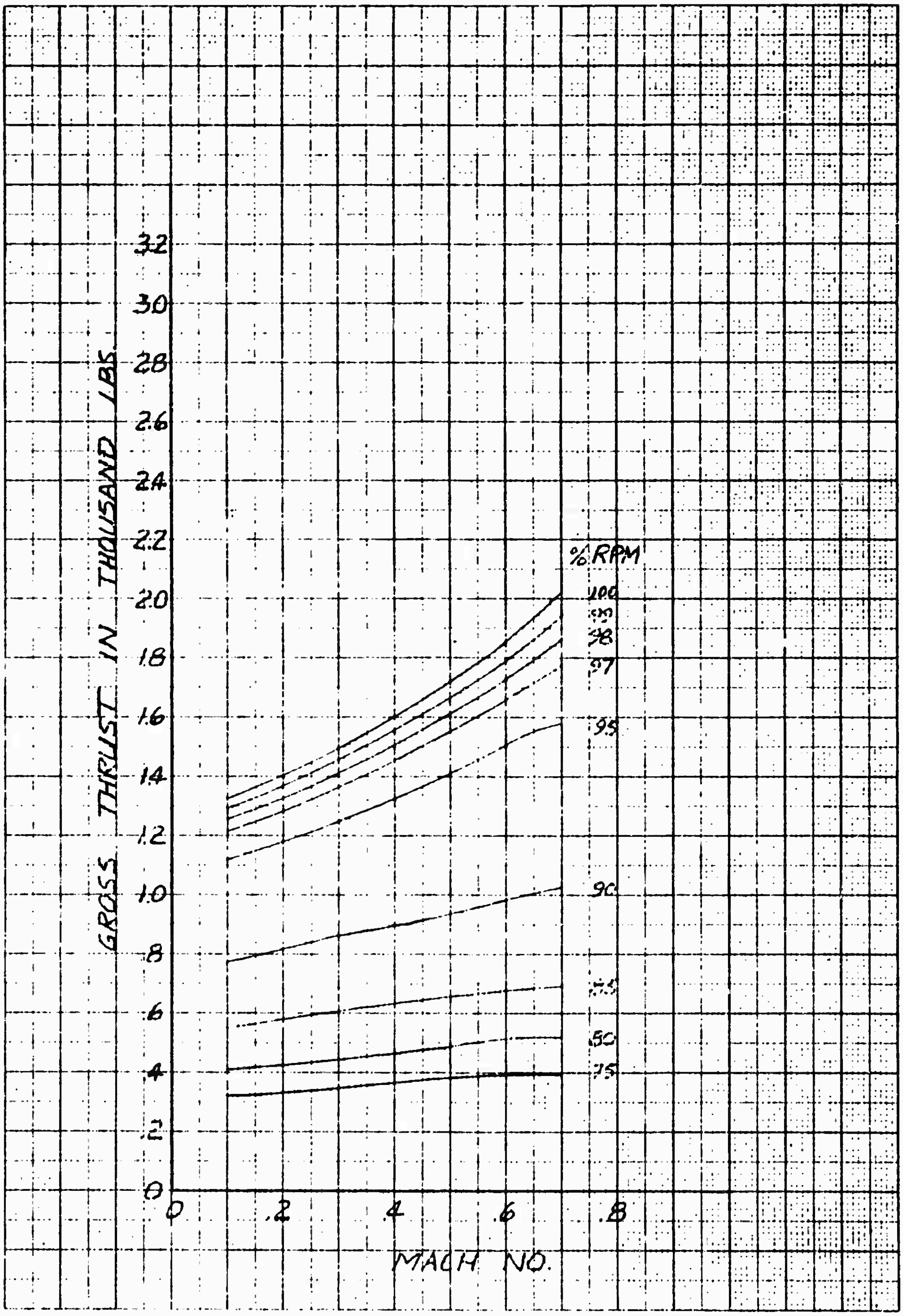

Figure 4.81 Gross Thrust vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Hot Day

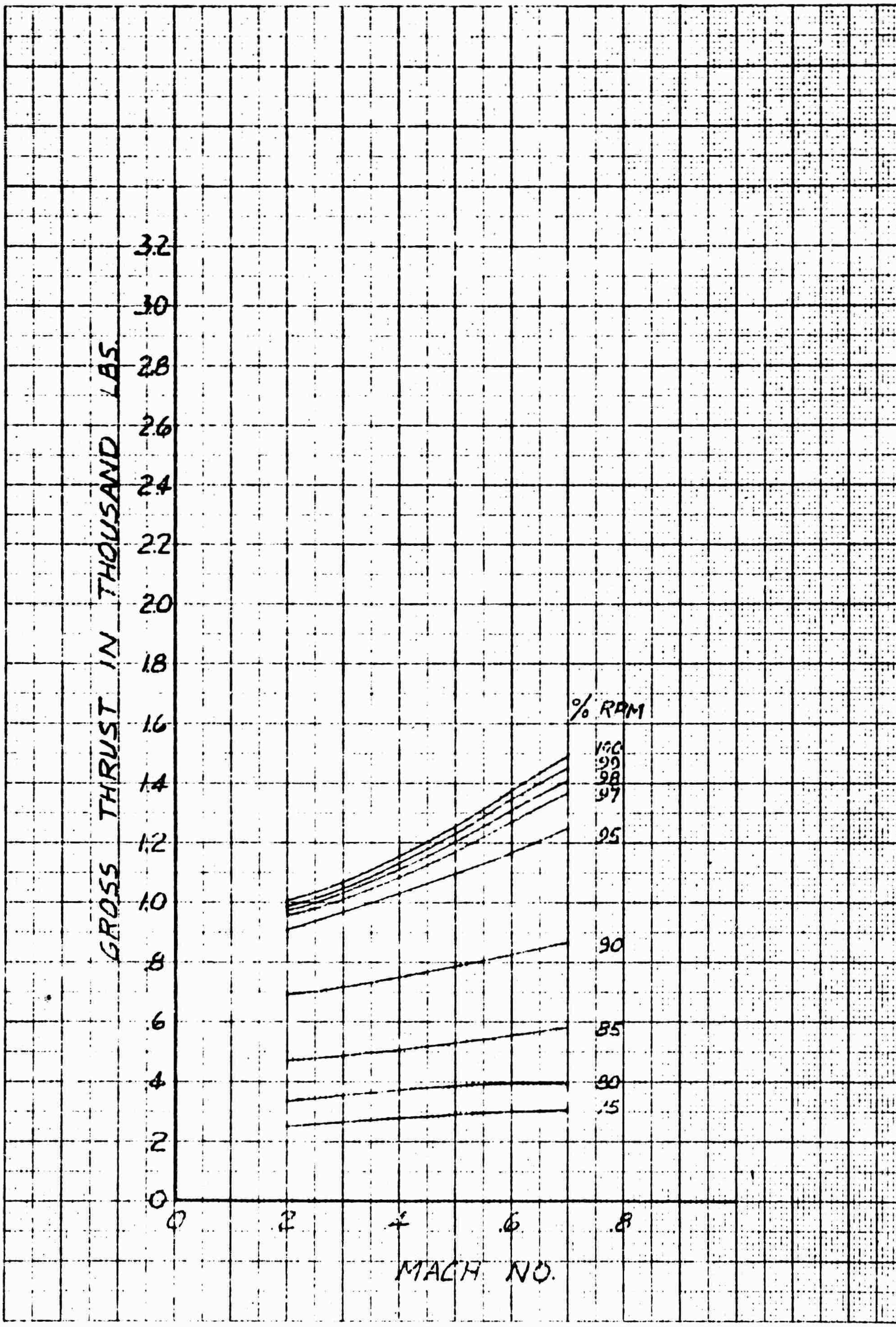

Figure 4.82 Gross Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Hot Day

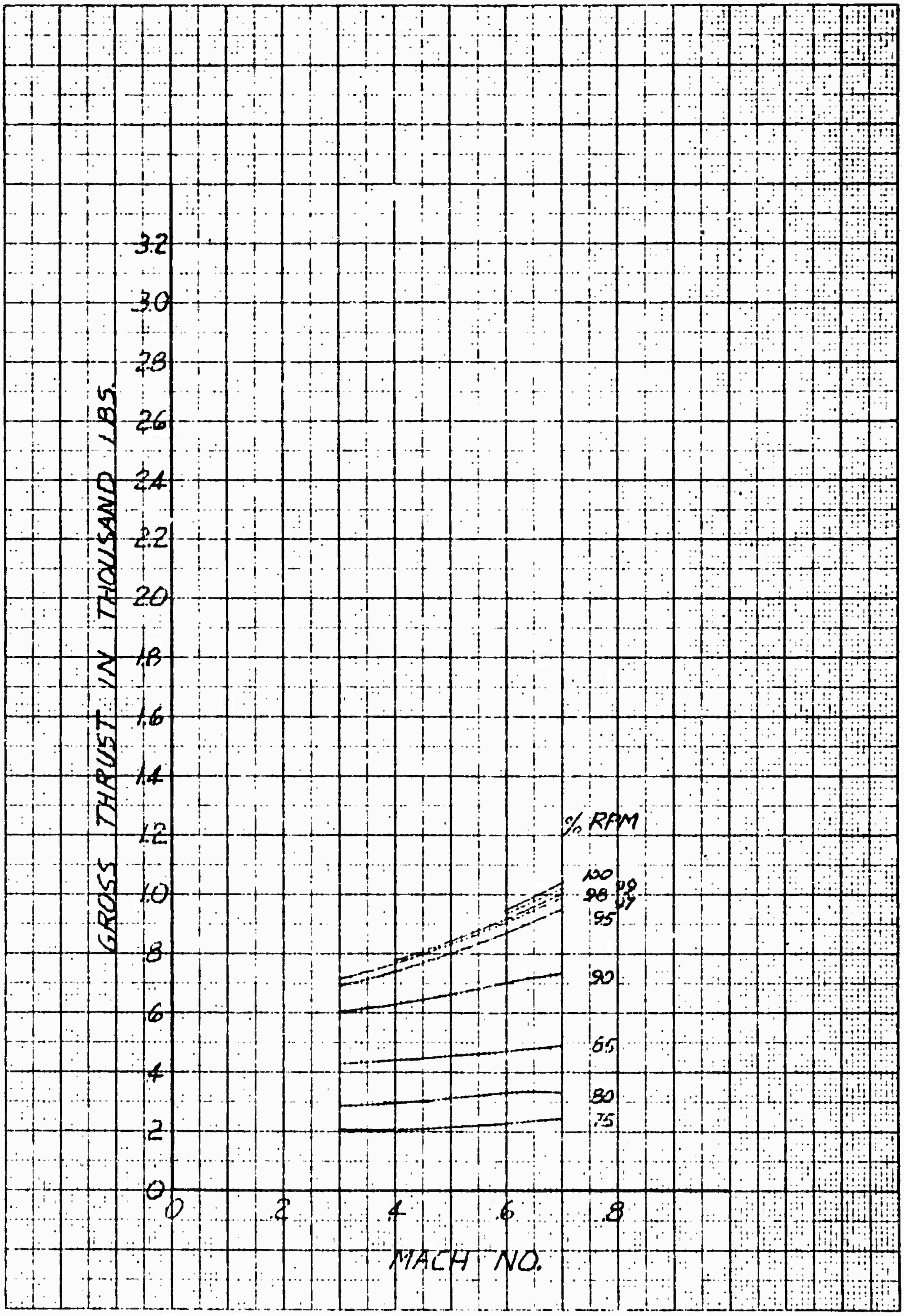

Figure 4.83 Gross Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Hot Day

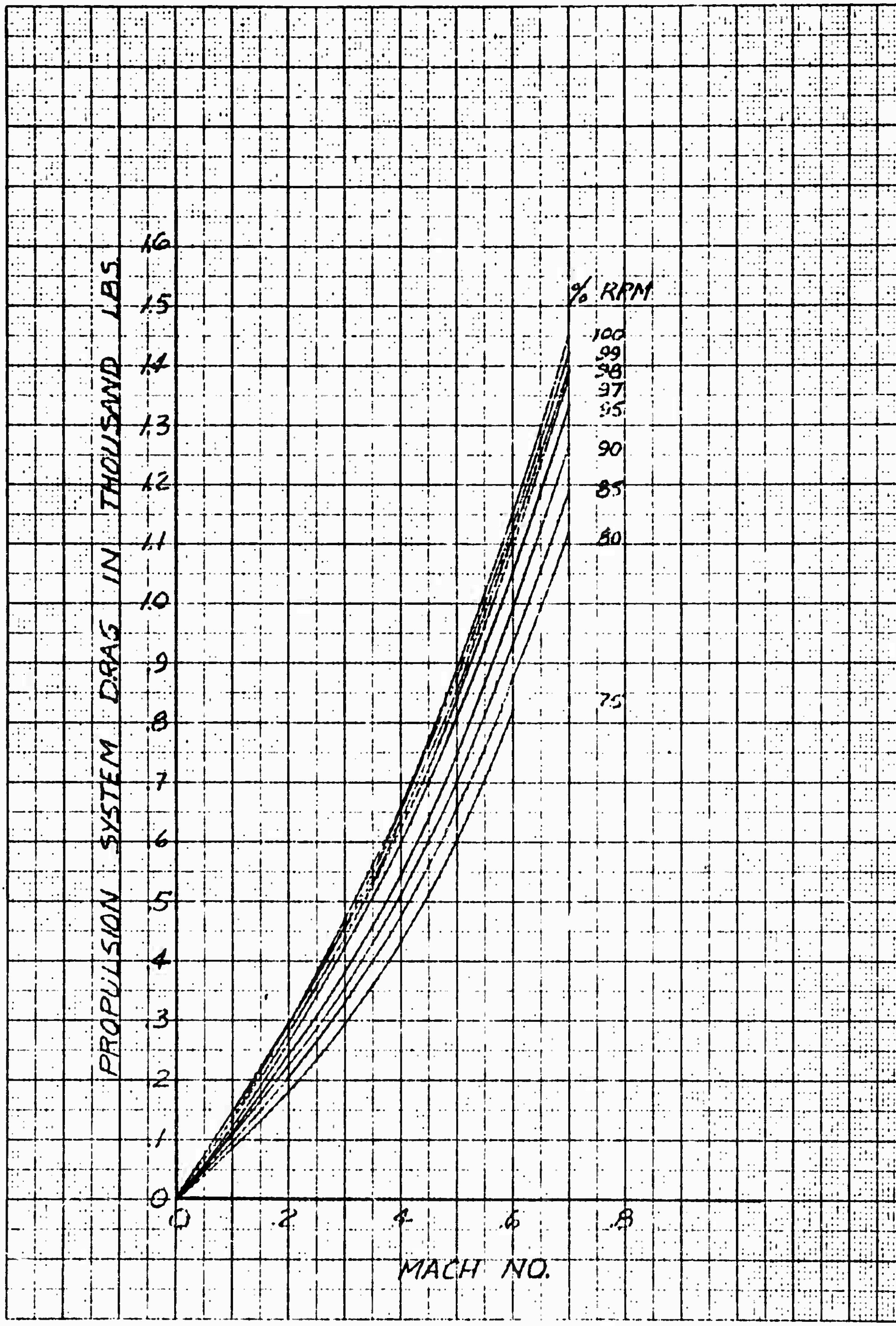

Figure 4.84 Propulsion System Drag vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Hot Day

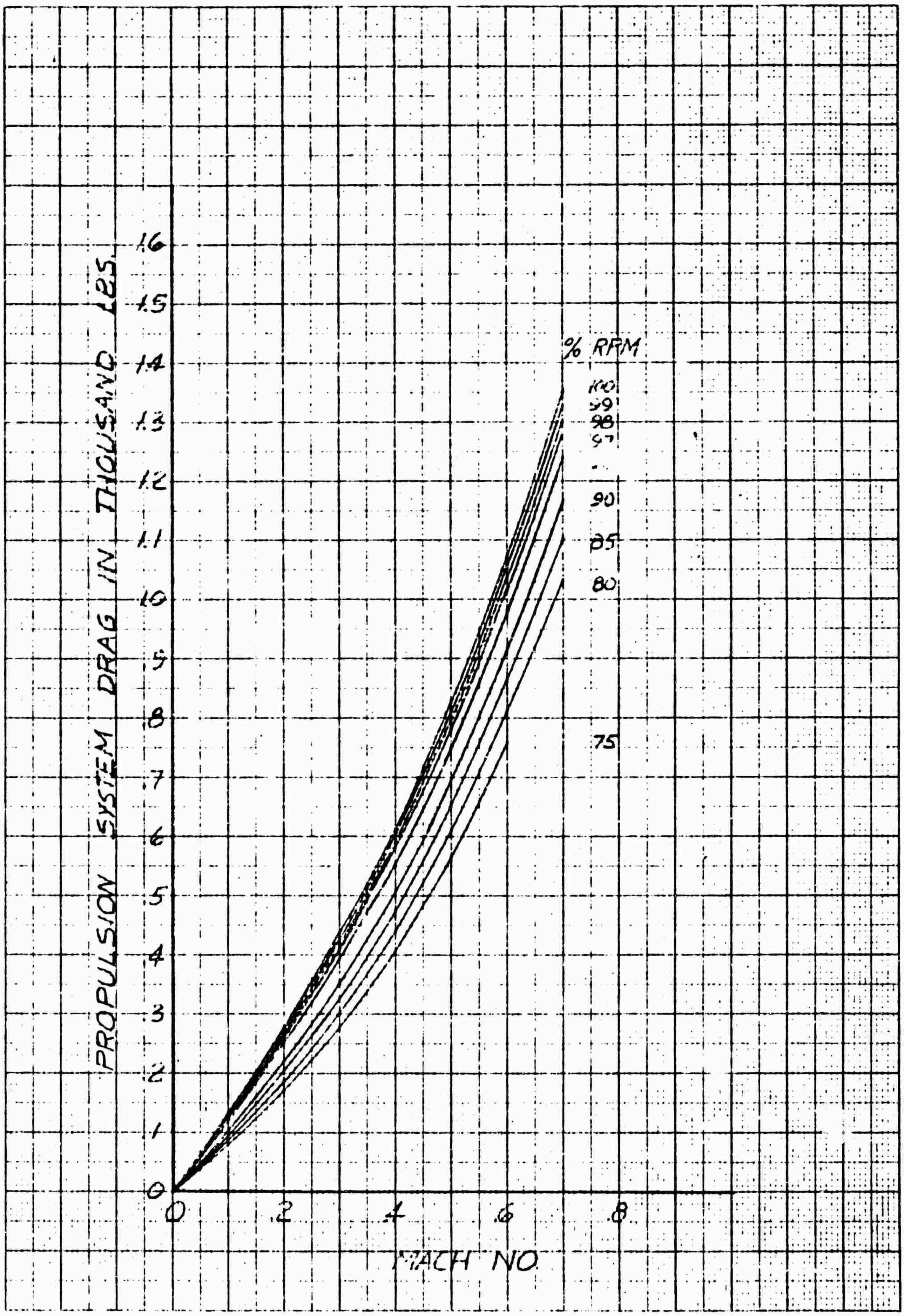

Figure 4.85 Propulsion System Drag vs Mach No. and % RPM; Altitude = 2500 ft., 1 Engine, Hot Day

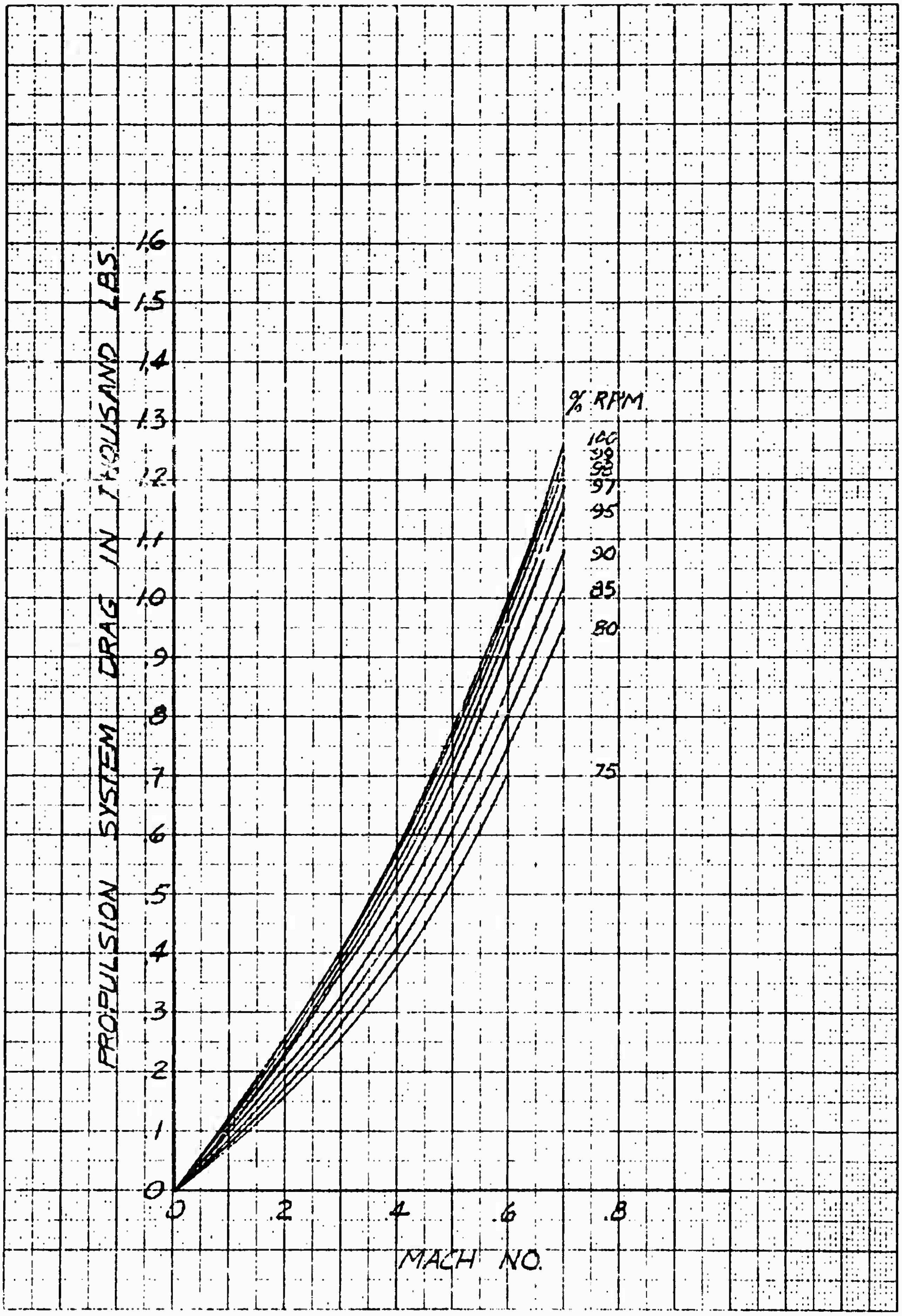

Figure 4.86 Propulsion System Drag vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Hot Day

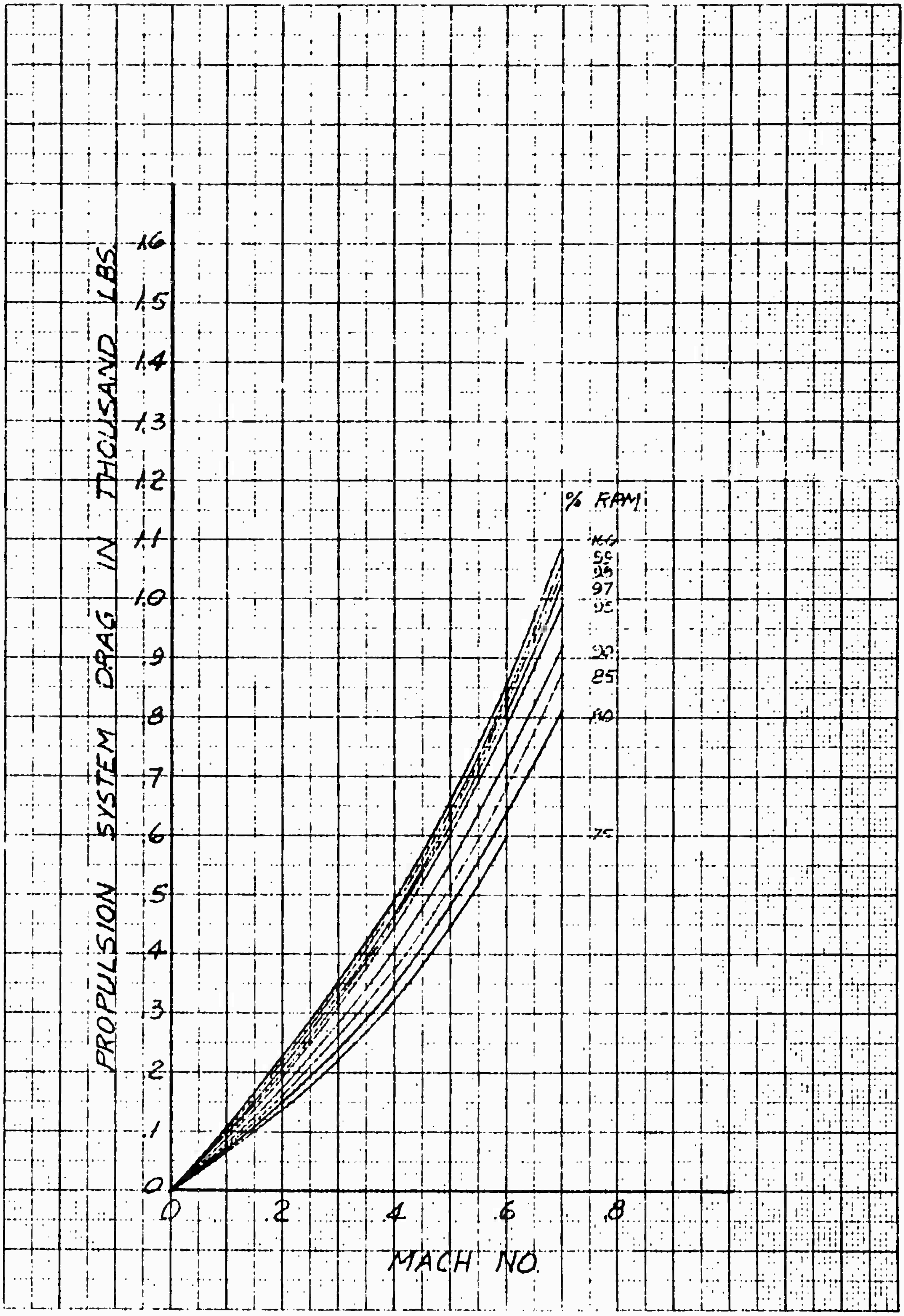

Figure 4.87 Propulsion System Drag vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Hot Day

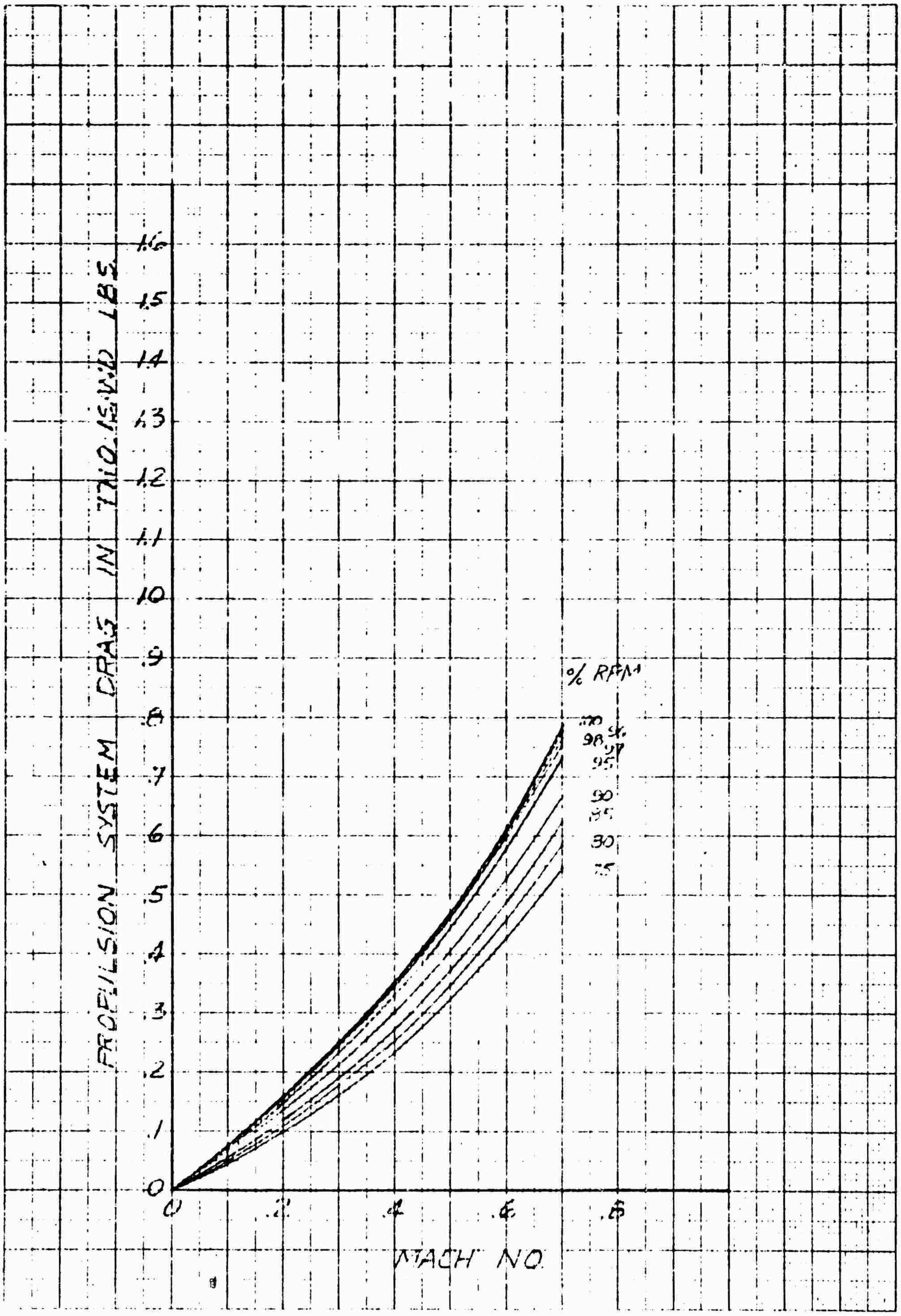

Figure 4.88 Propulsion System Drag vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Hot Day

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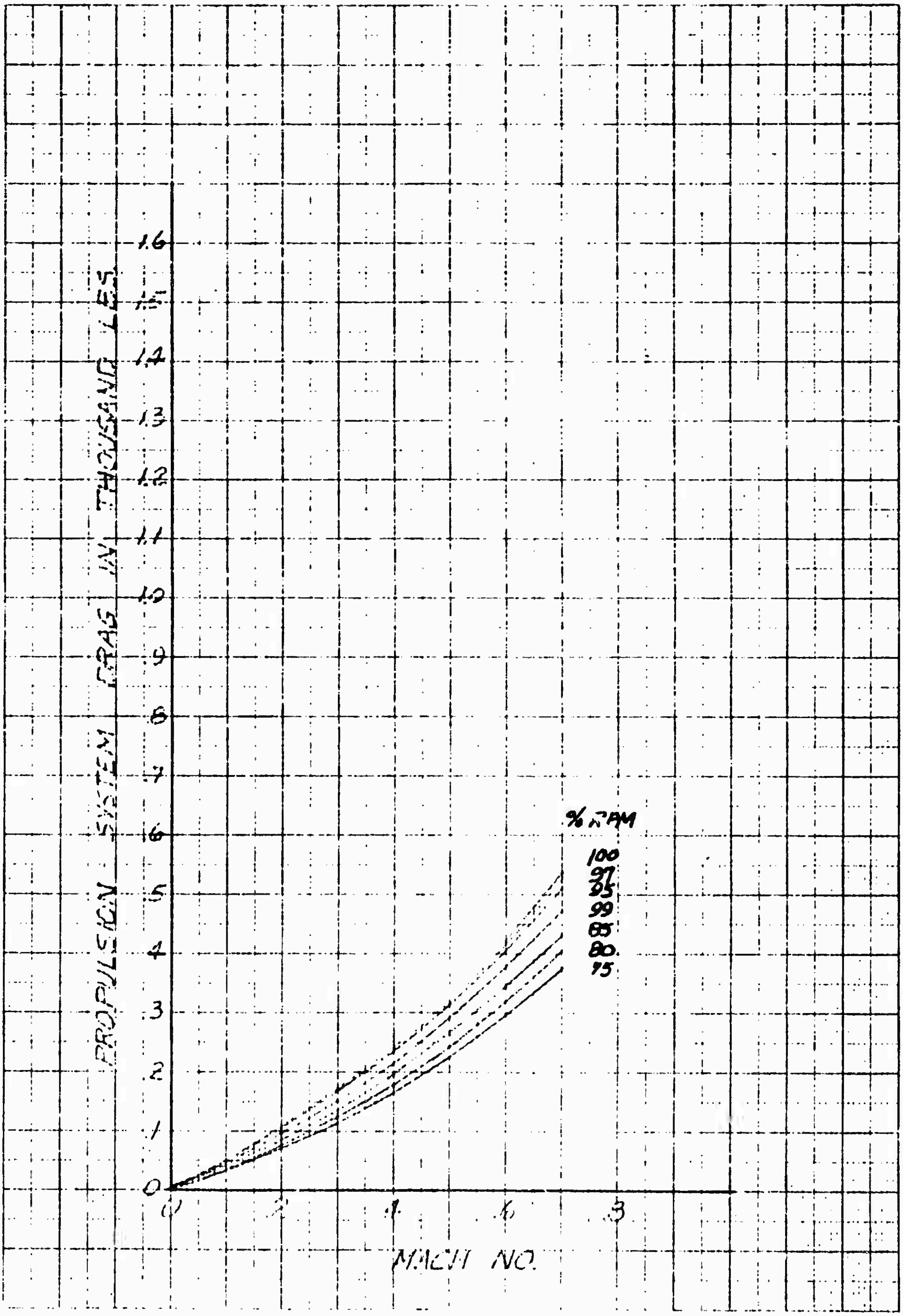

Figure 4.89 Propulsion System Drag vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Hot Day

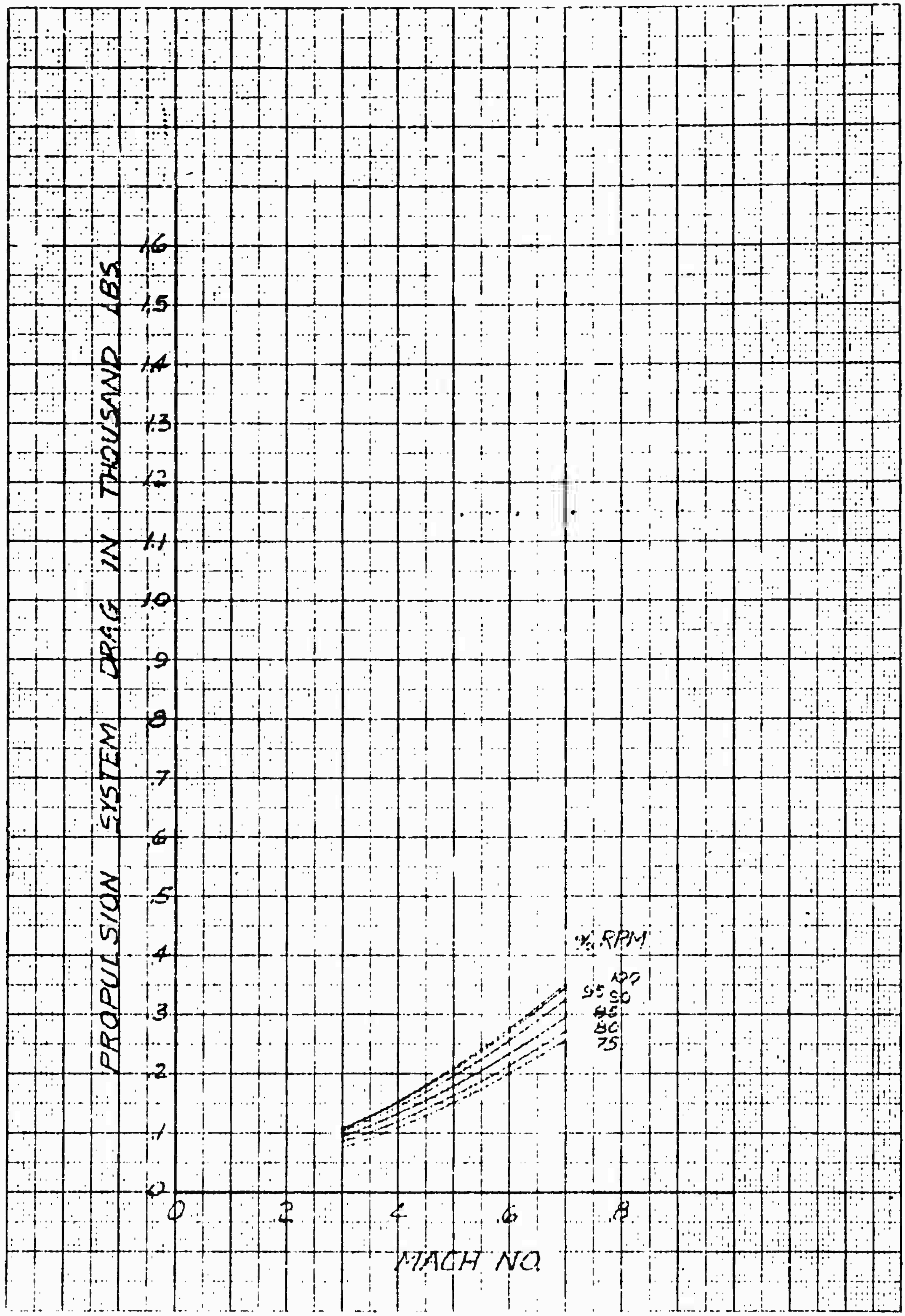

Figure 4.90 Propulsion System Drag vs Mach No. and % RPM; Altitude = 40,000 ft. 1 Engine, Hot Day

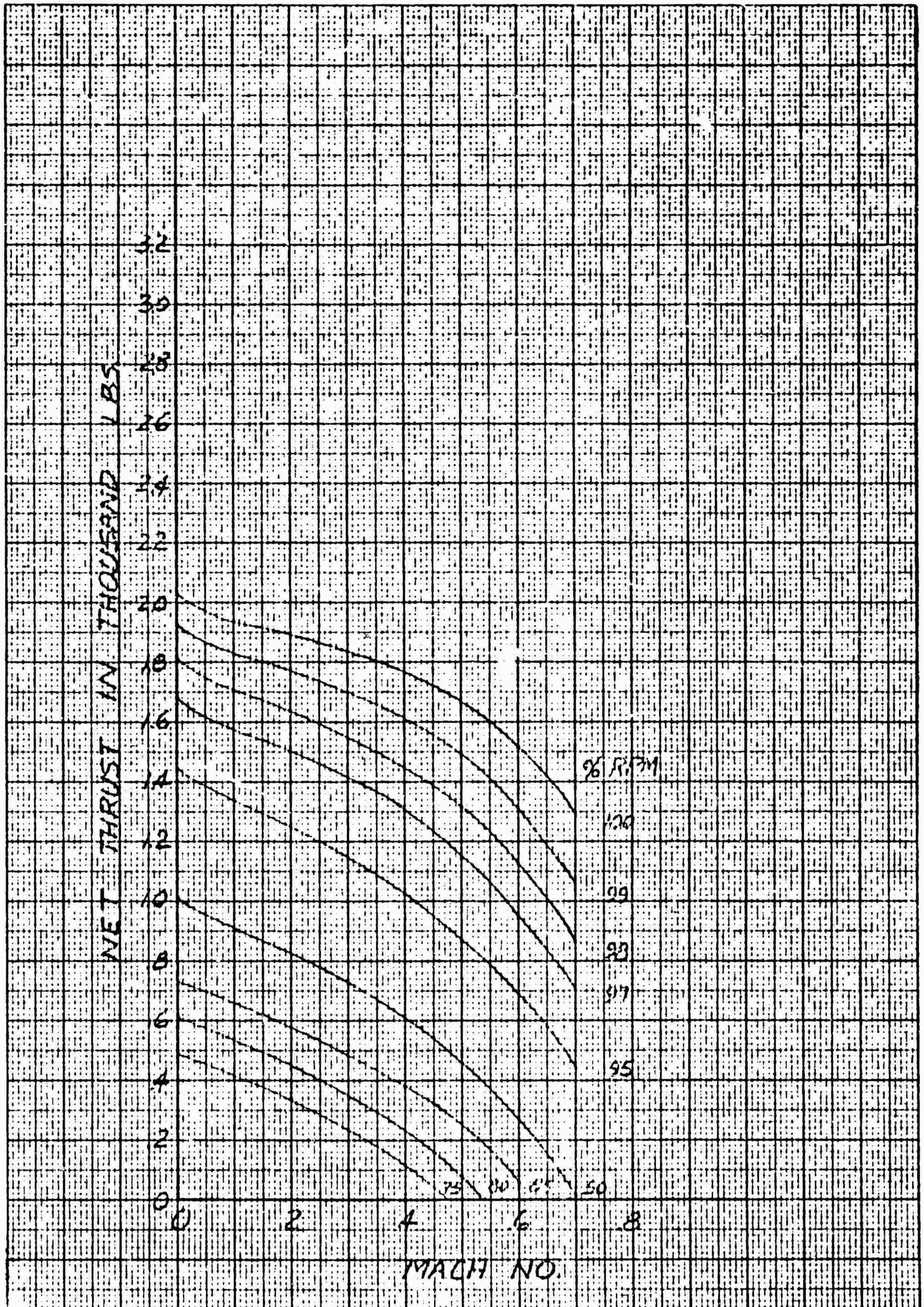

Figure 4.91 Net Thrust vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Hot Day

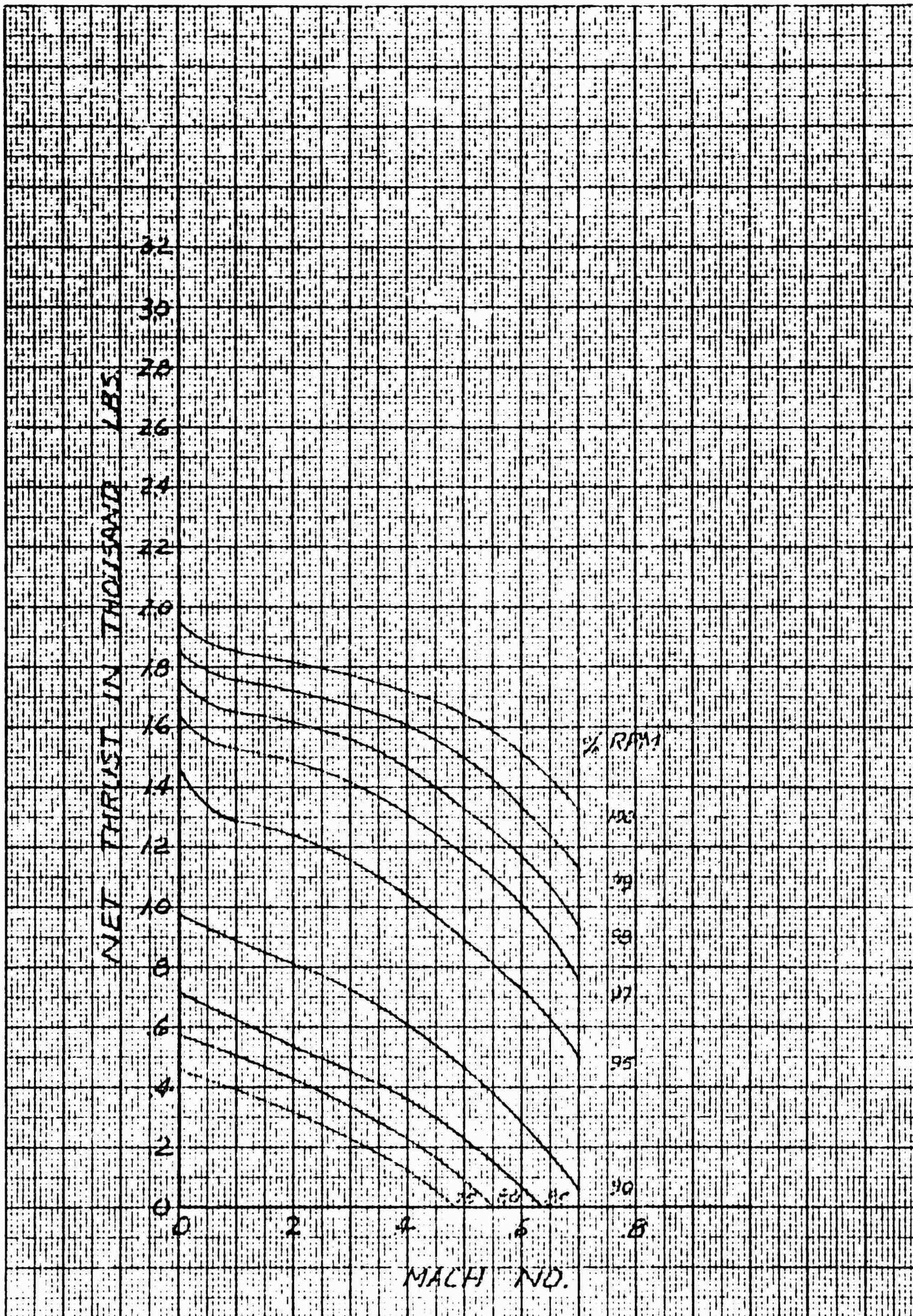

Figure 4.92 Net Thrust vs Mach No. and % RPM; Altitude = 2500 ft., 1 Engine, Hot Day

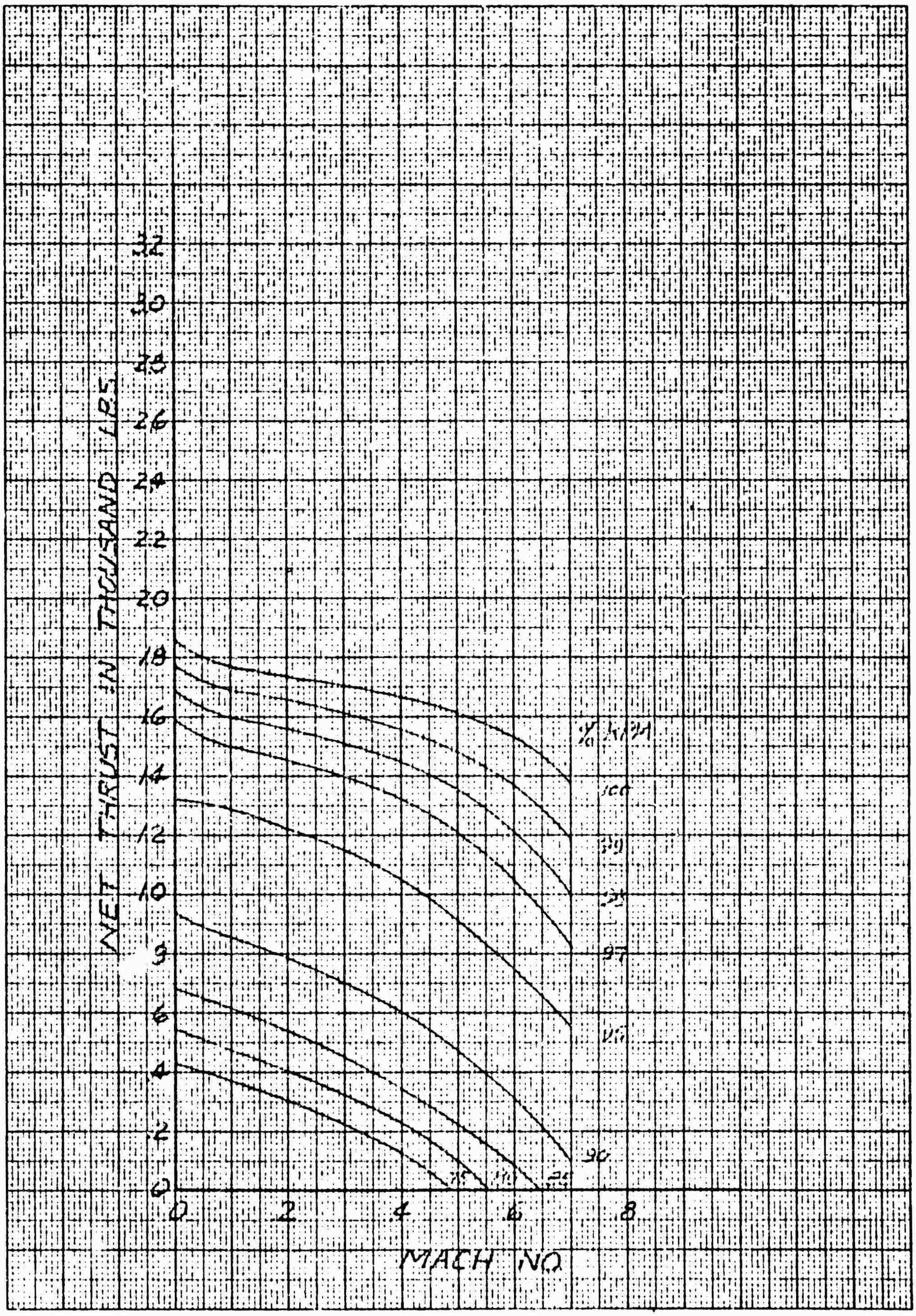

Figure 4.93 Net Thrust vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Hot Day

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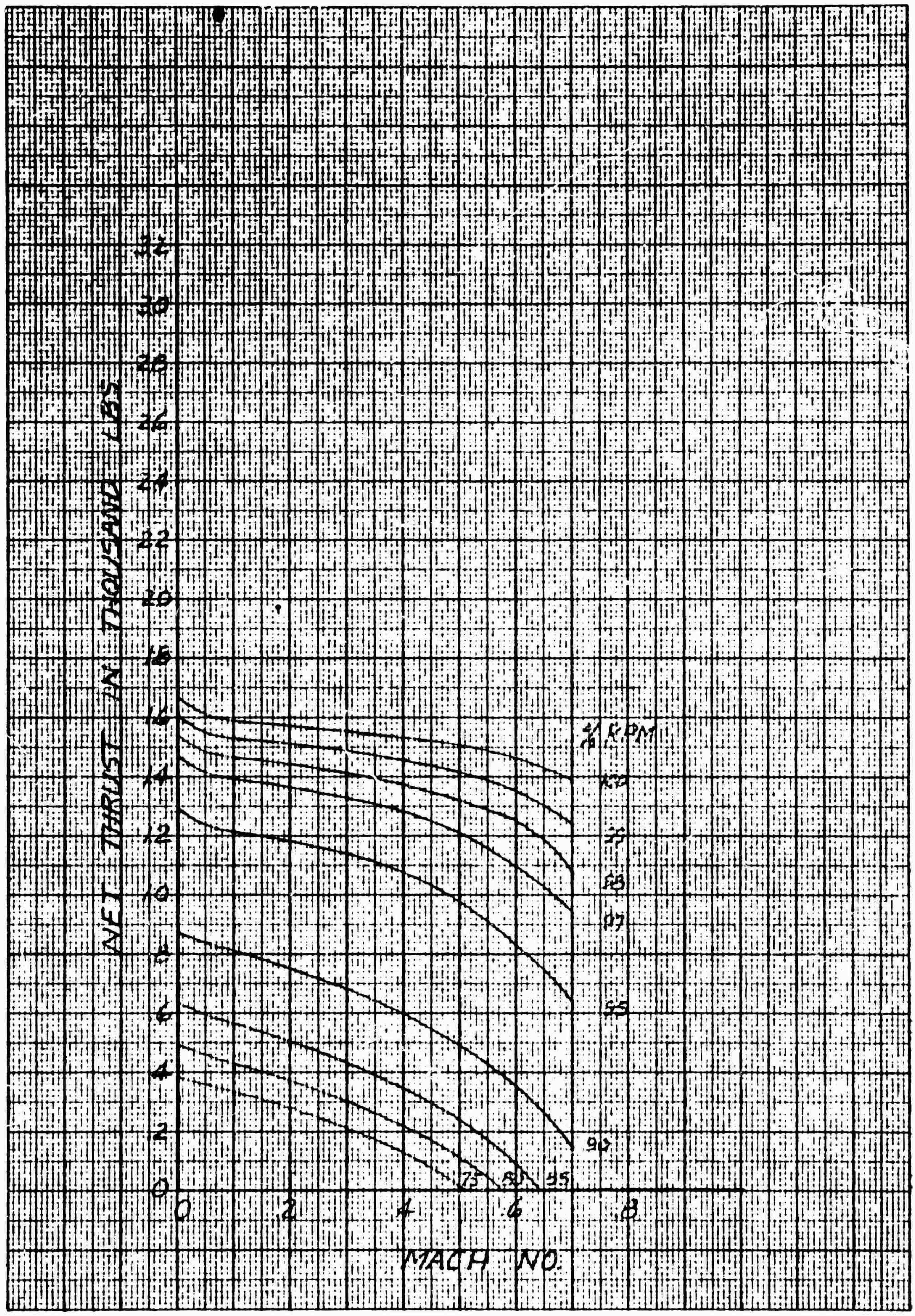

Figure 4.94 Net Thrust vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Hot Day

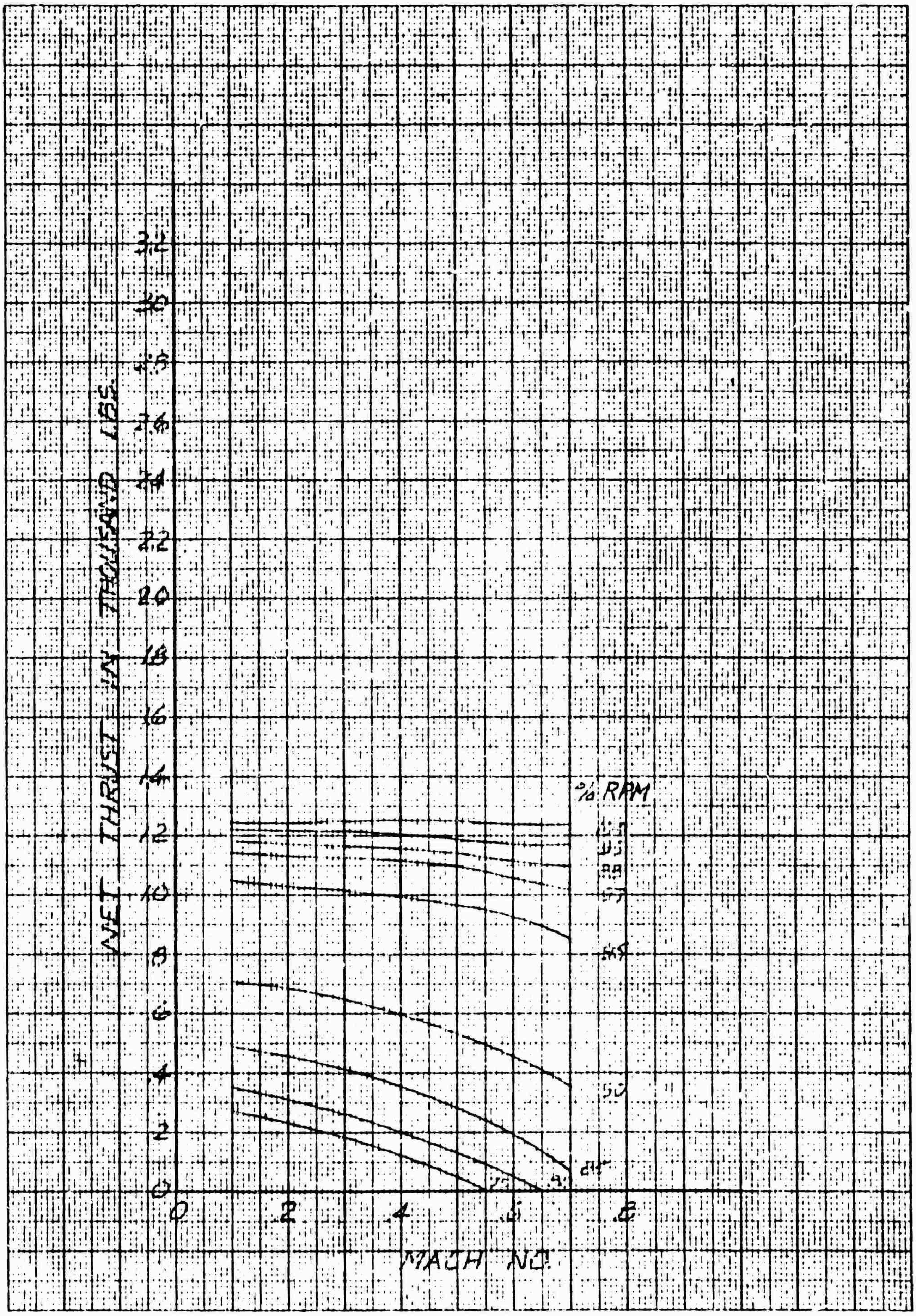

Figure 4.95 Net Thrust vs Mach No. and % RPM; Altitude= 20,000 ft., 1 Engine, Hot Day

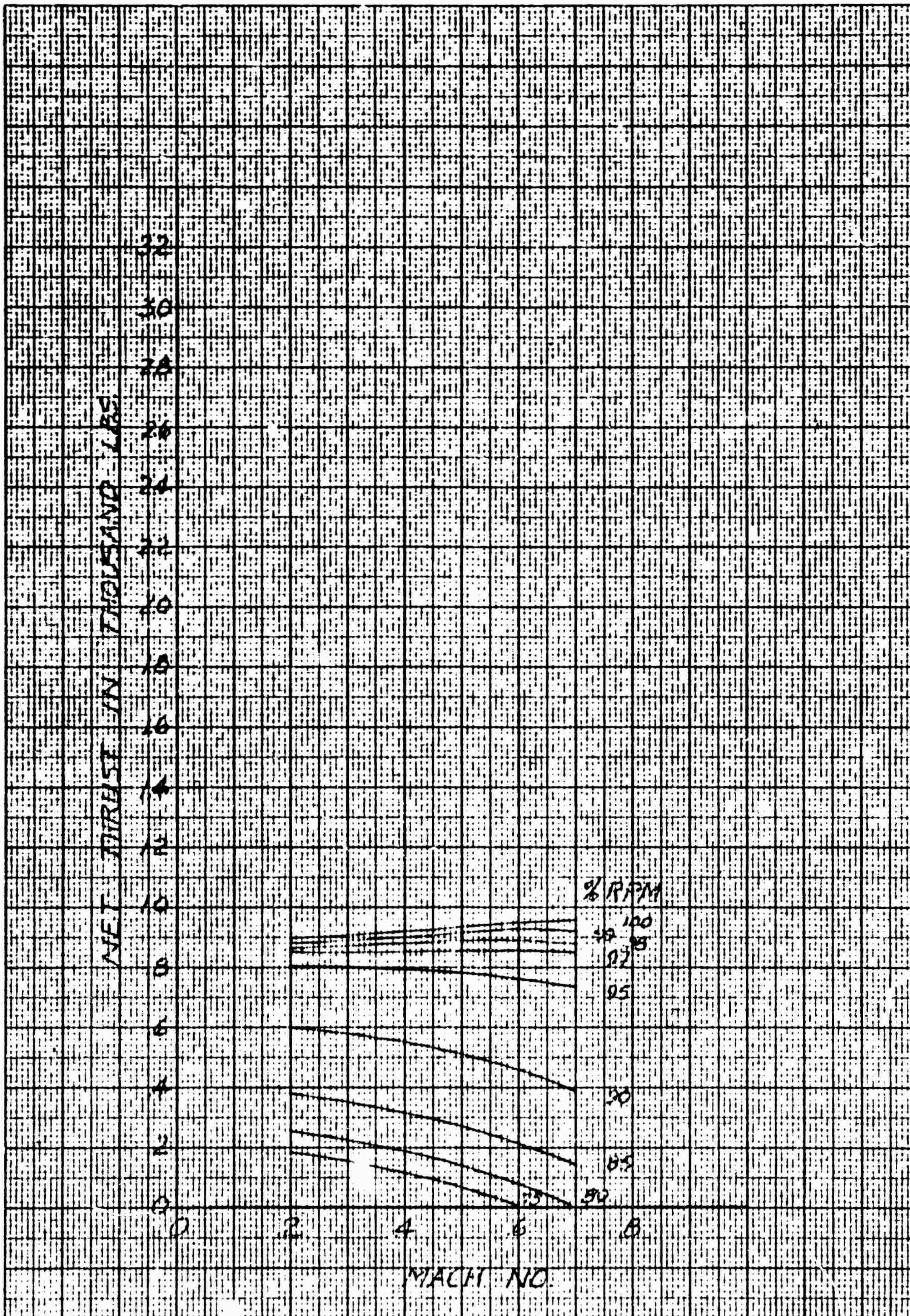

Figure 4.96 Net Thrust vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Hot Day

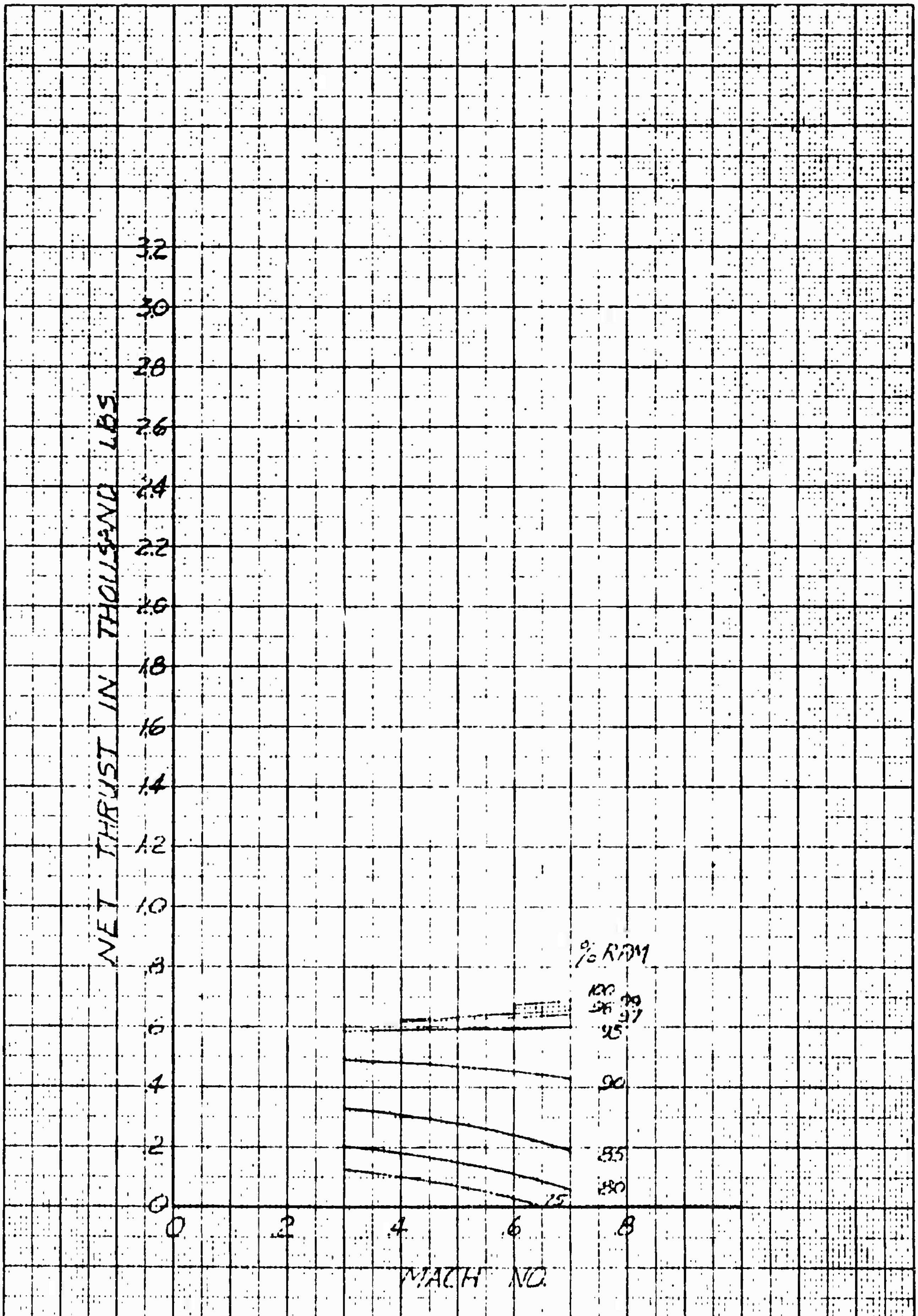

Figure 4.97 Net Thrust vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Hot Day

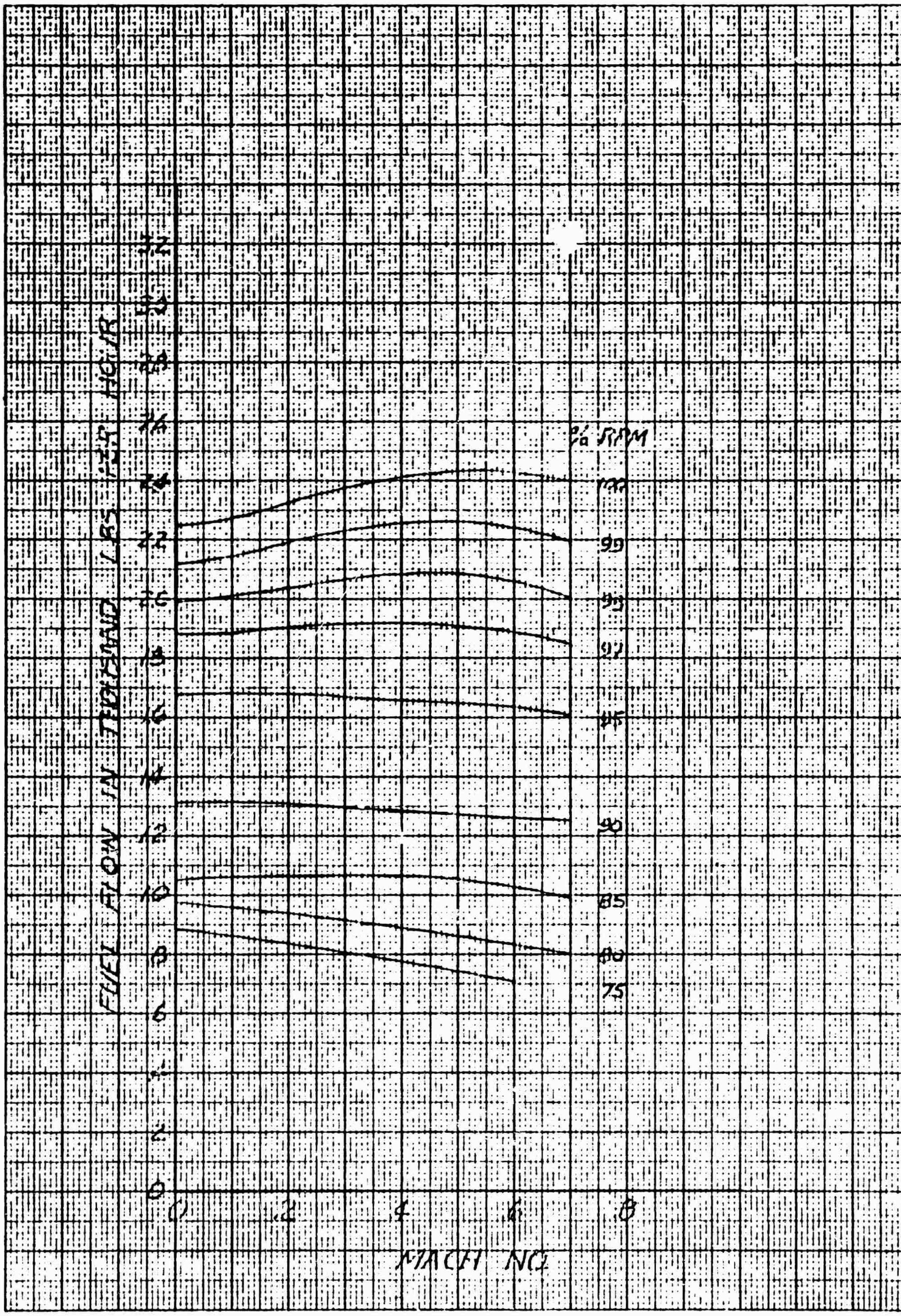

Figure 4.98 Fuel Flow vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Hot Day

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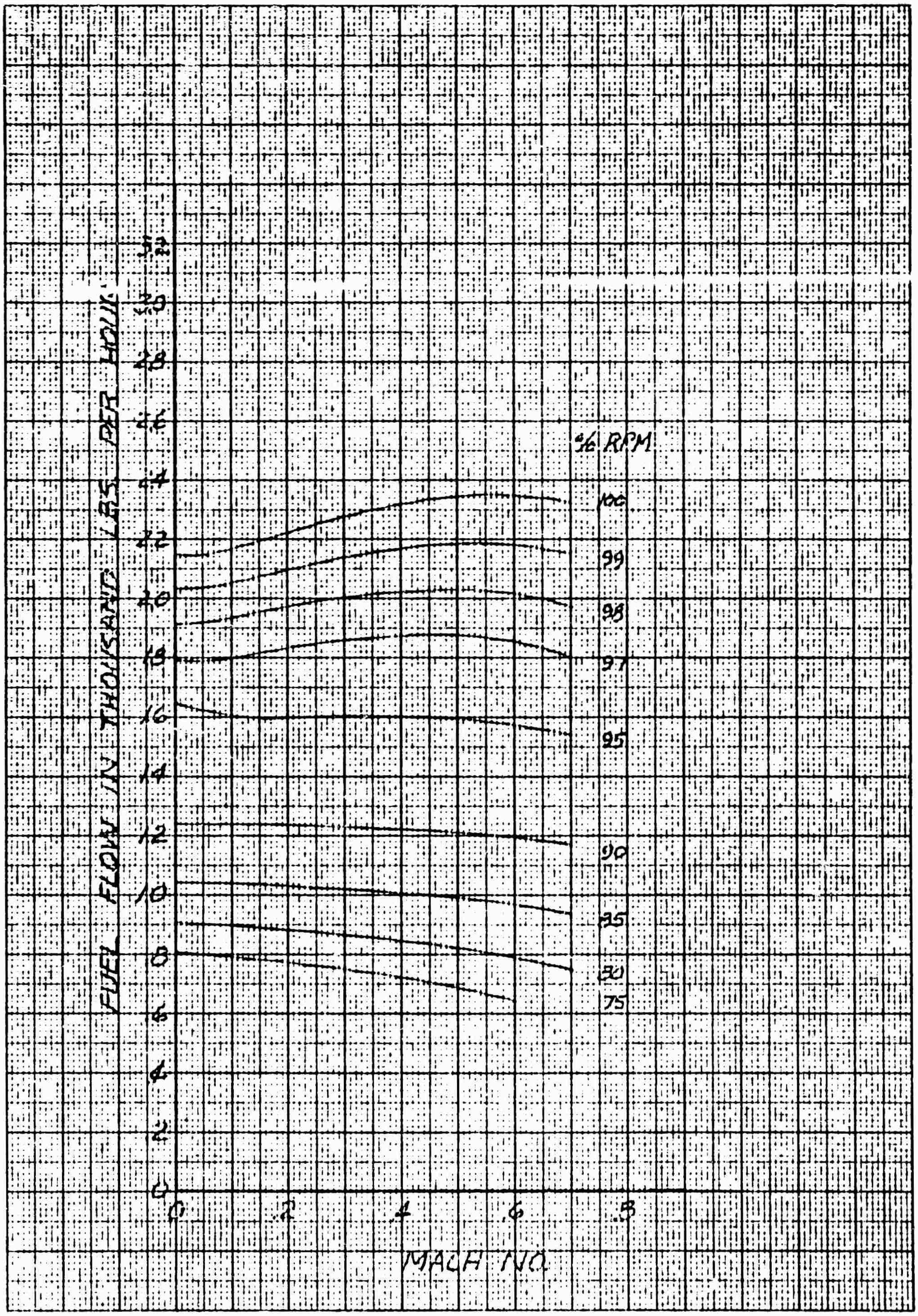

Figure 4.99 Fuel Flow vs Mach No. and % RPM; Altitude = 2500 ft., 1 Engine, Hot Day

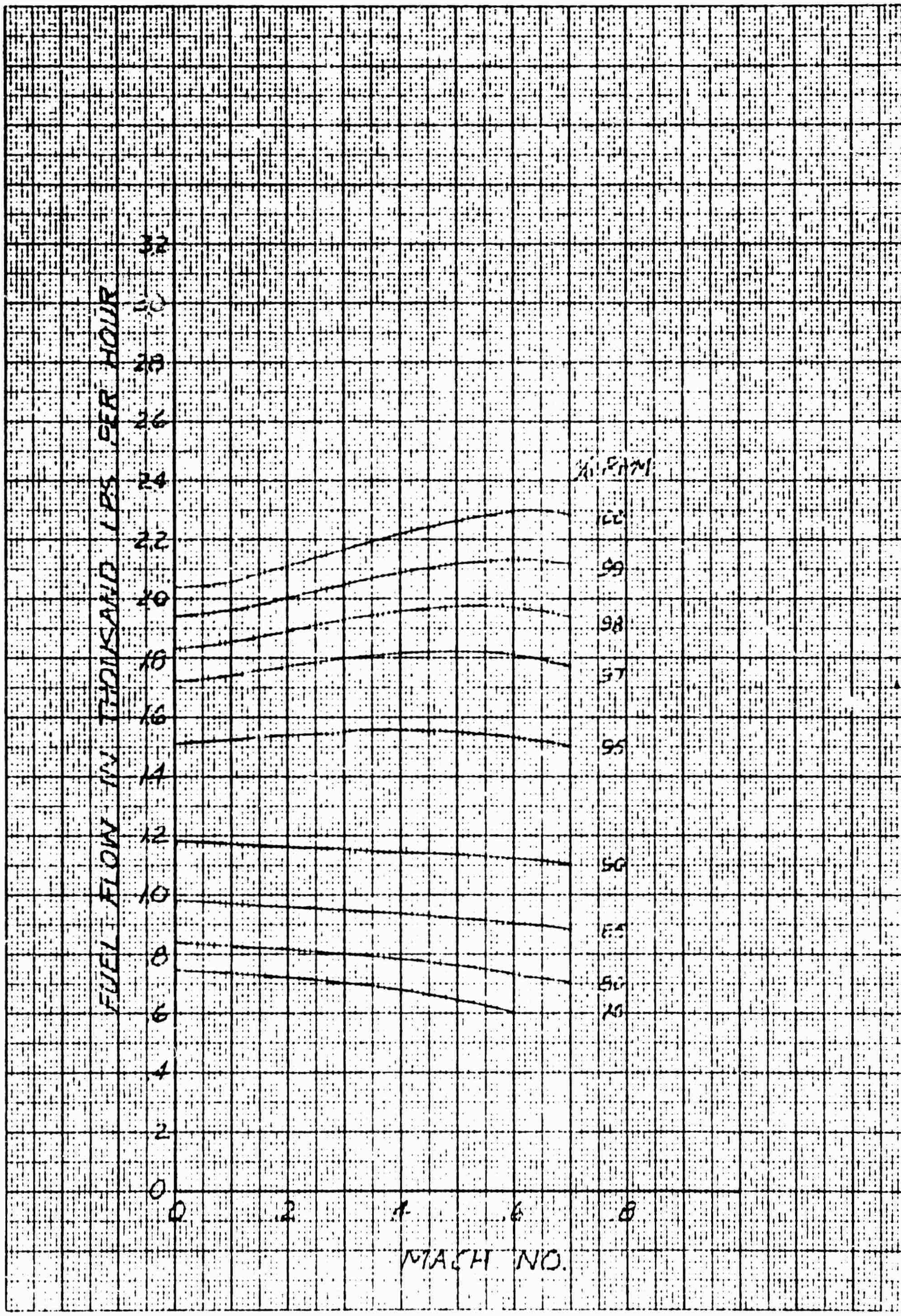

Figure 4.100 Fuel Flow vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Hot Day

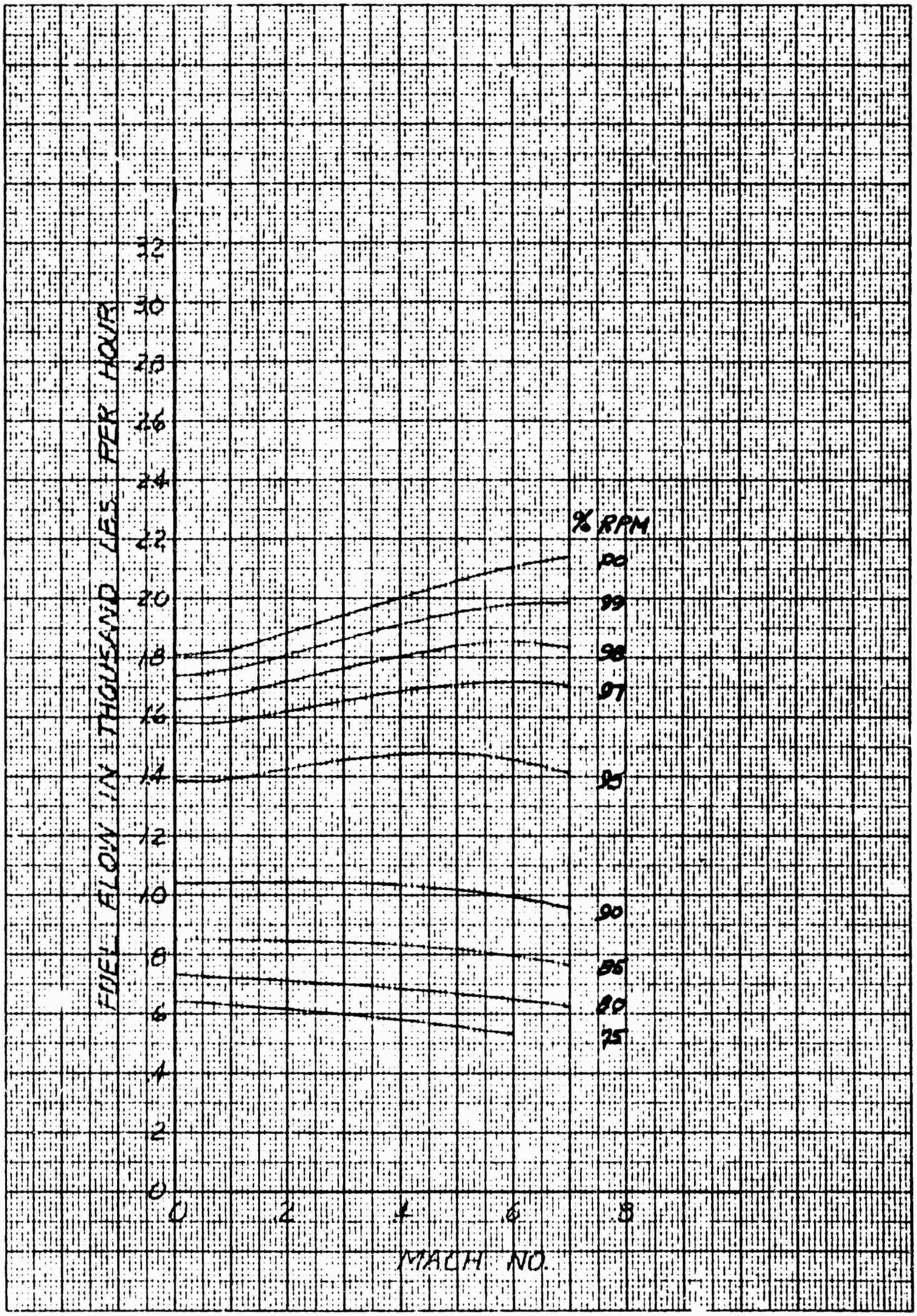

Figure 4.101 Fuel Flow vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Hot Day

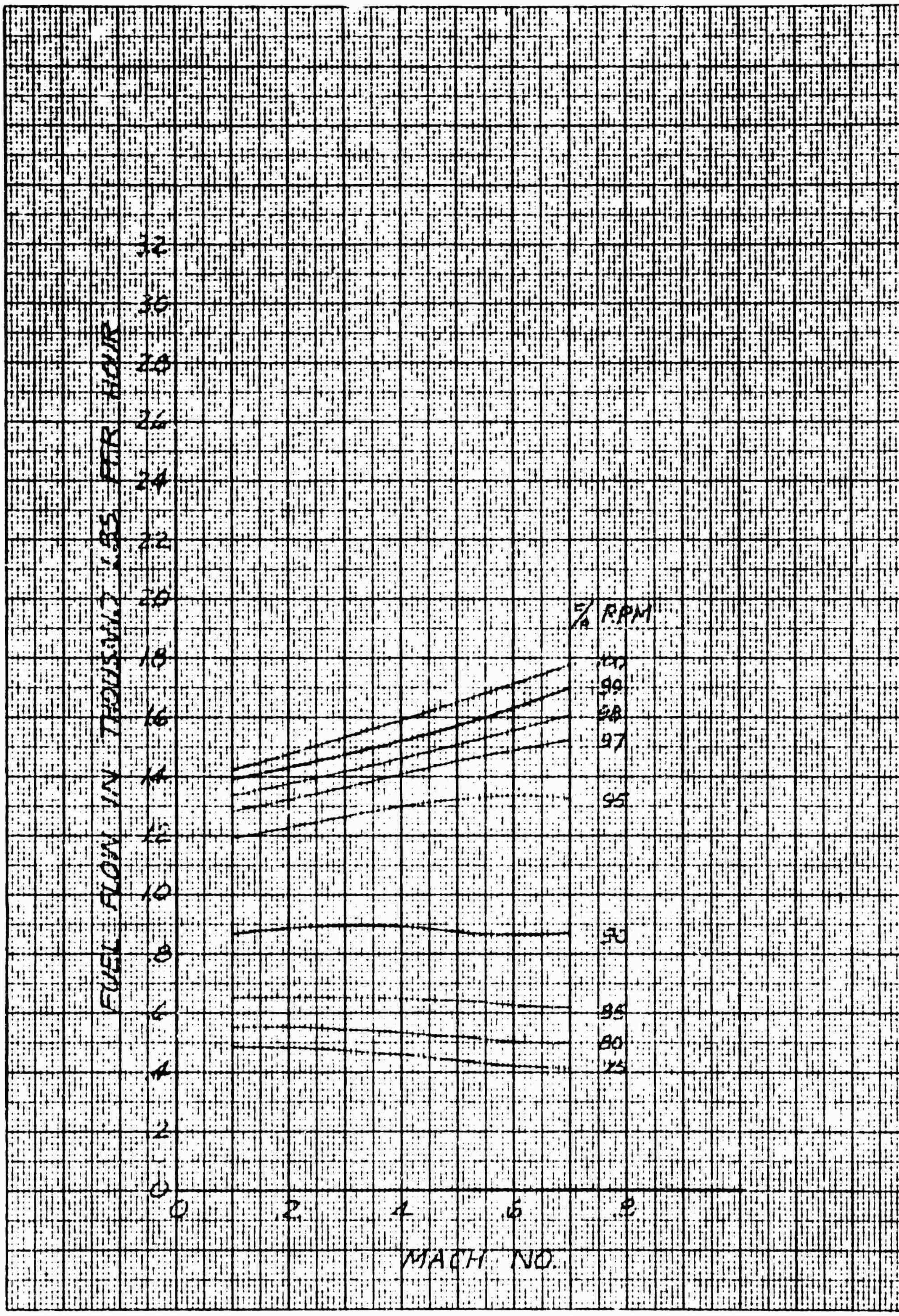

Figure 4.102 Fuel Flow vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Hot Day

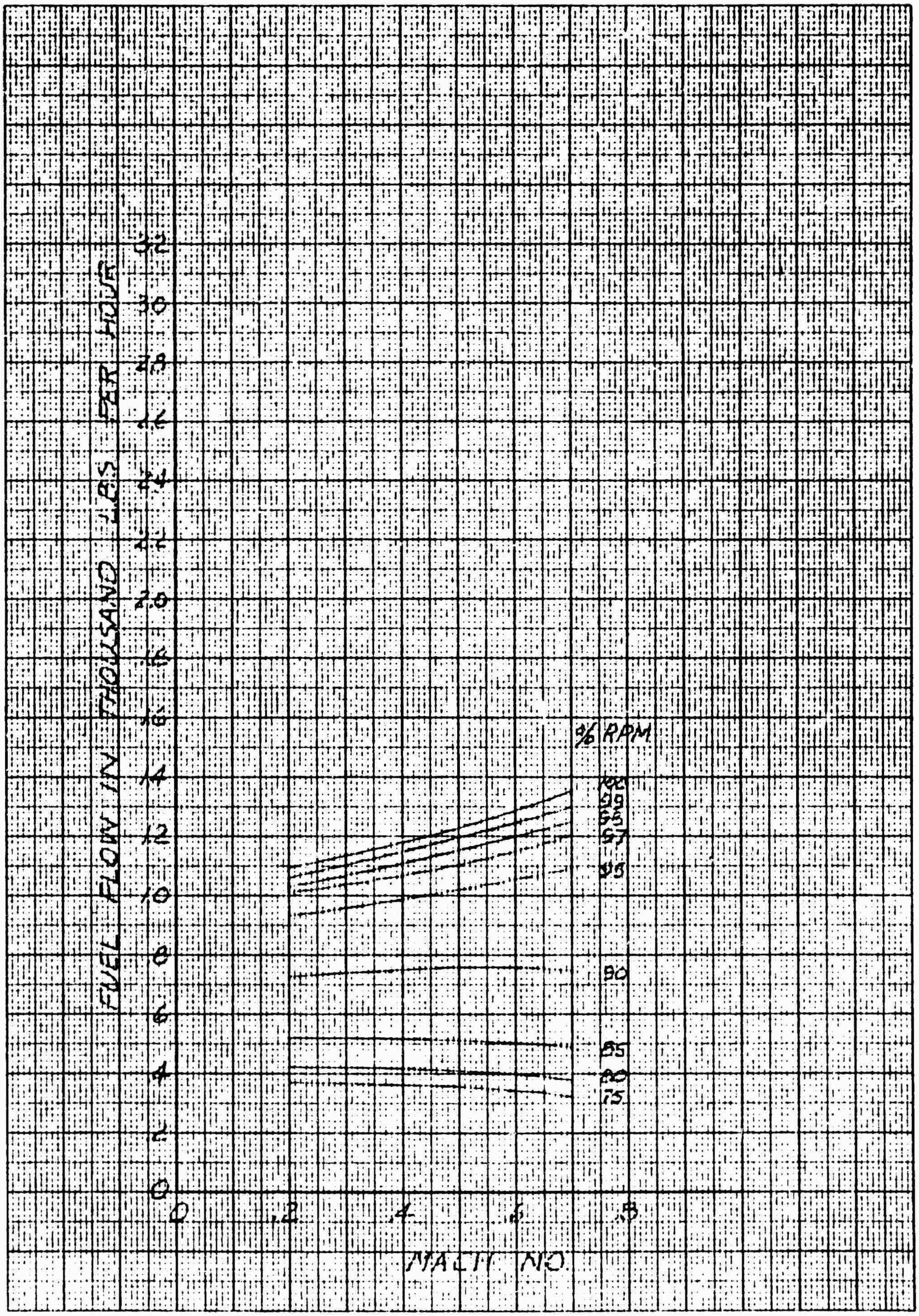

Figure 4.103 Fuel Flow vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Hot Day

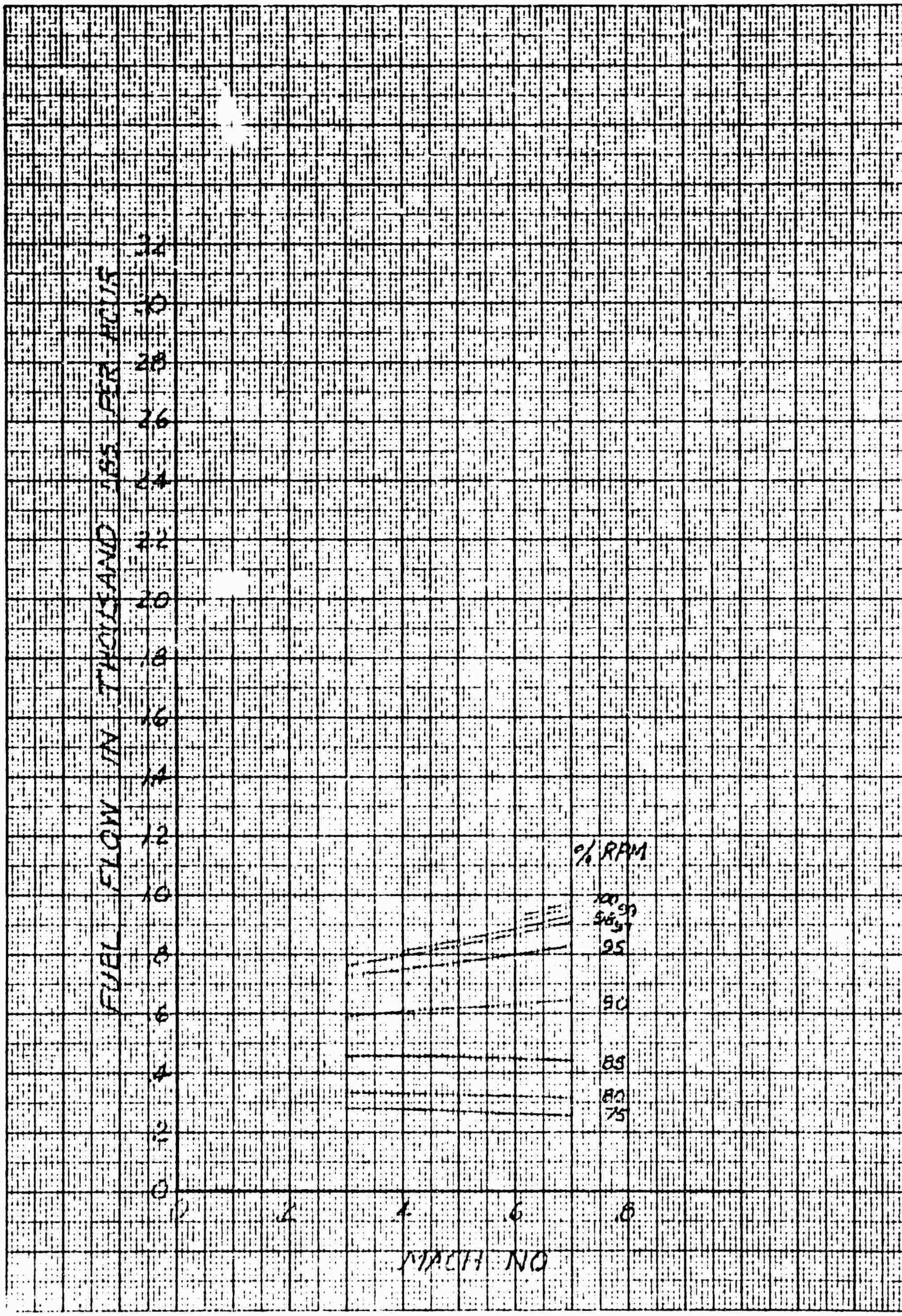

Figure 4.104 Fuel Flow vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Hot Day

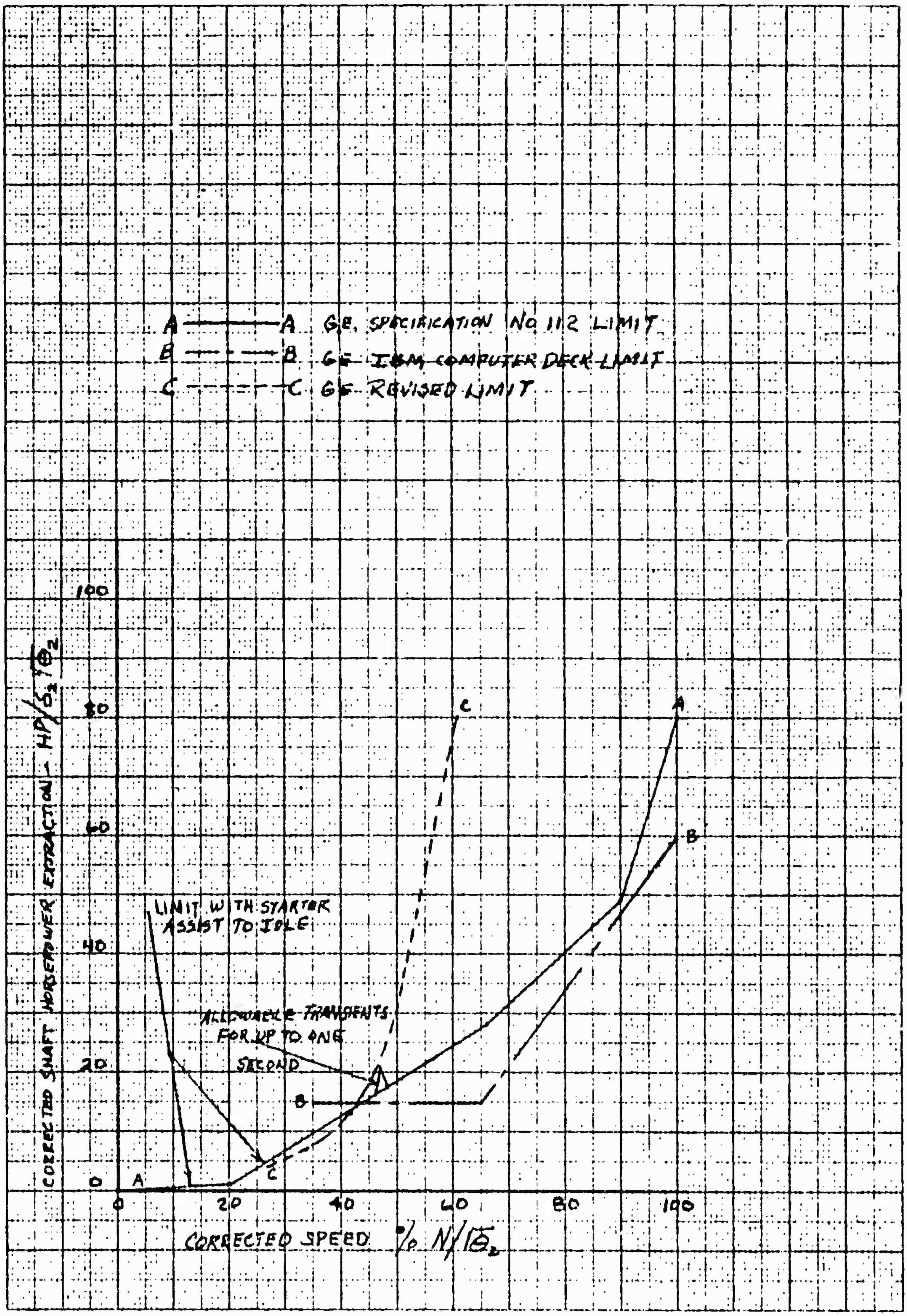

Figure 4.105 Corrected Shaft Horsepower Extraction vs Corrected % Speed

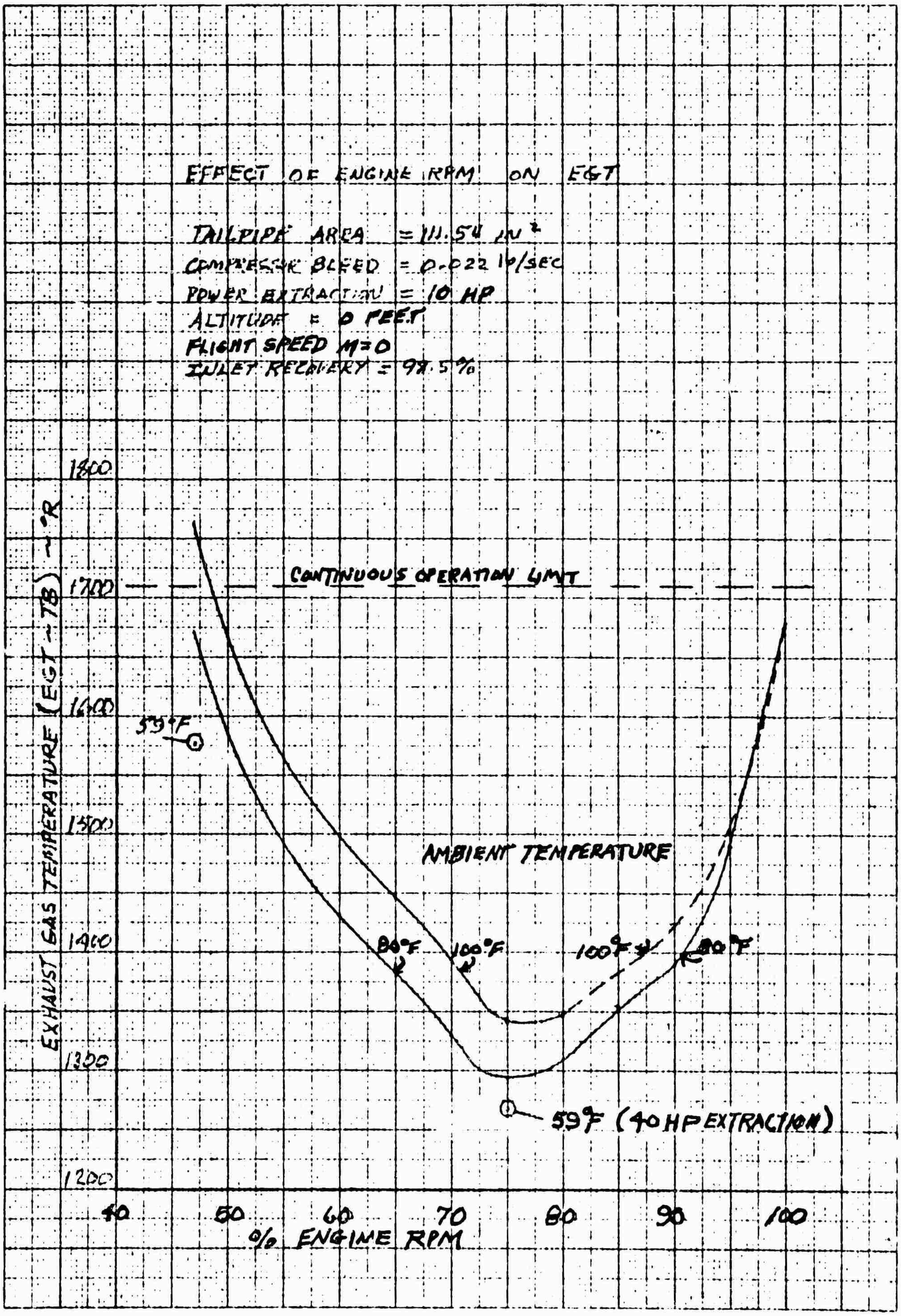

Figure 4.106 Effect of Engine RPM on Exhaust Gas Temperature

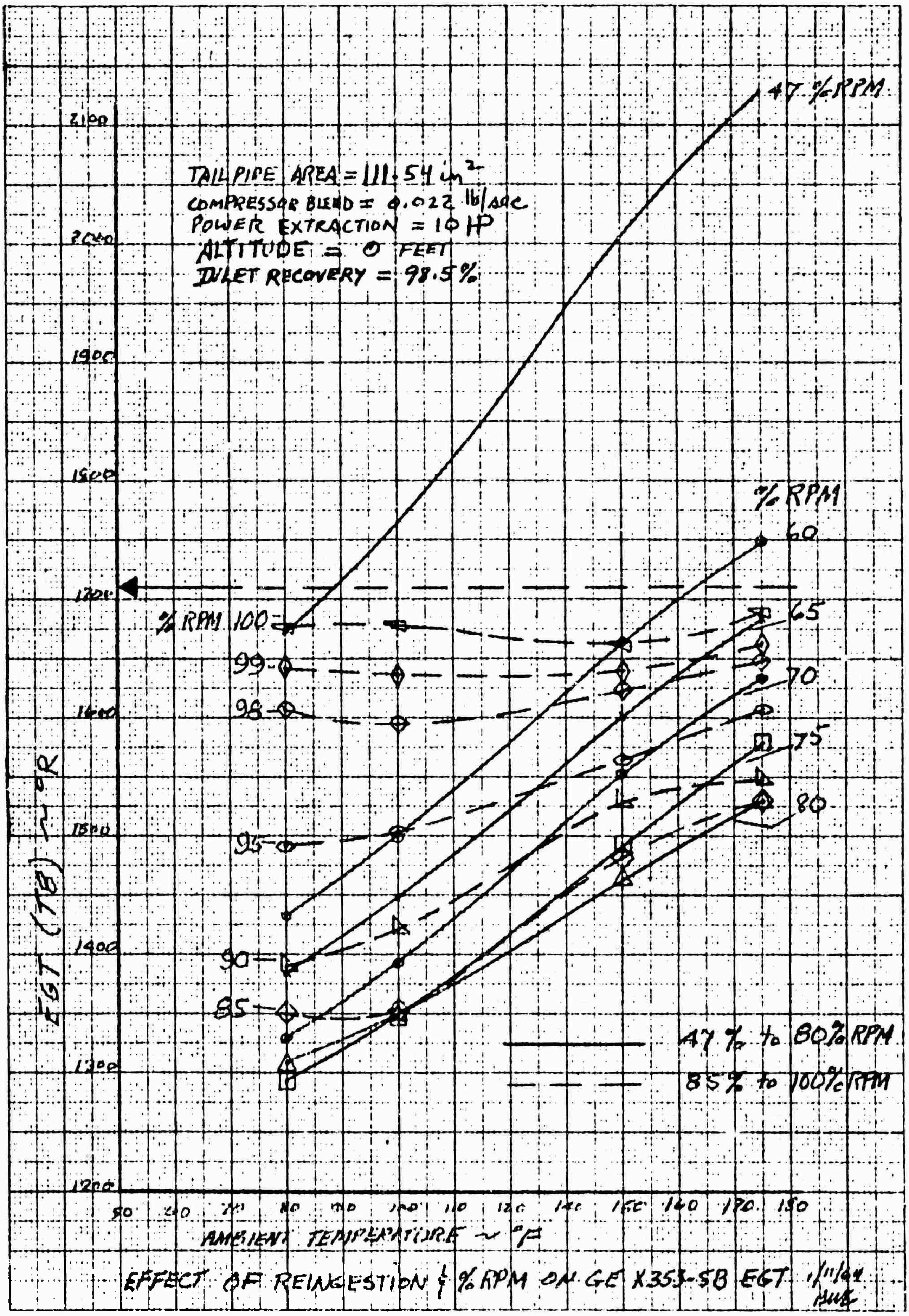

Figure 4.107 Effect of Reingestion and % RPM on Exhaust Gas Temperature

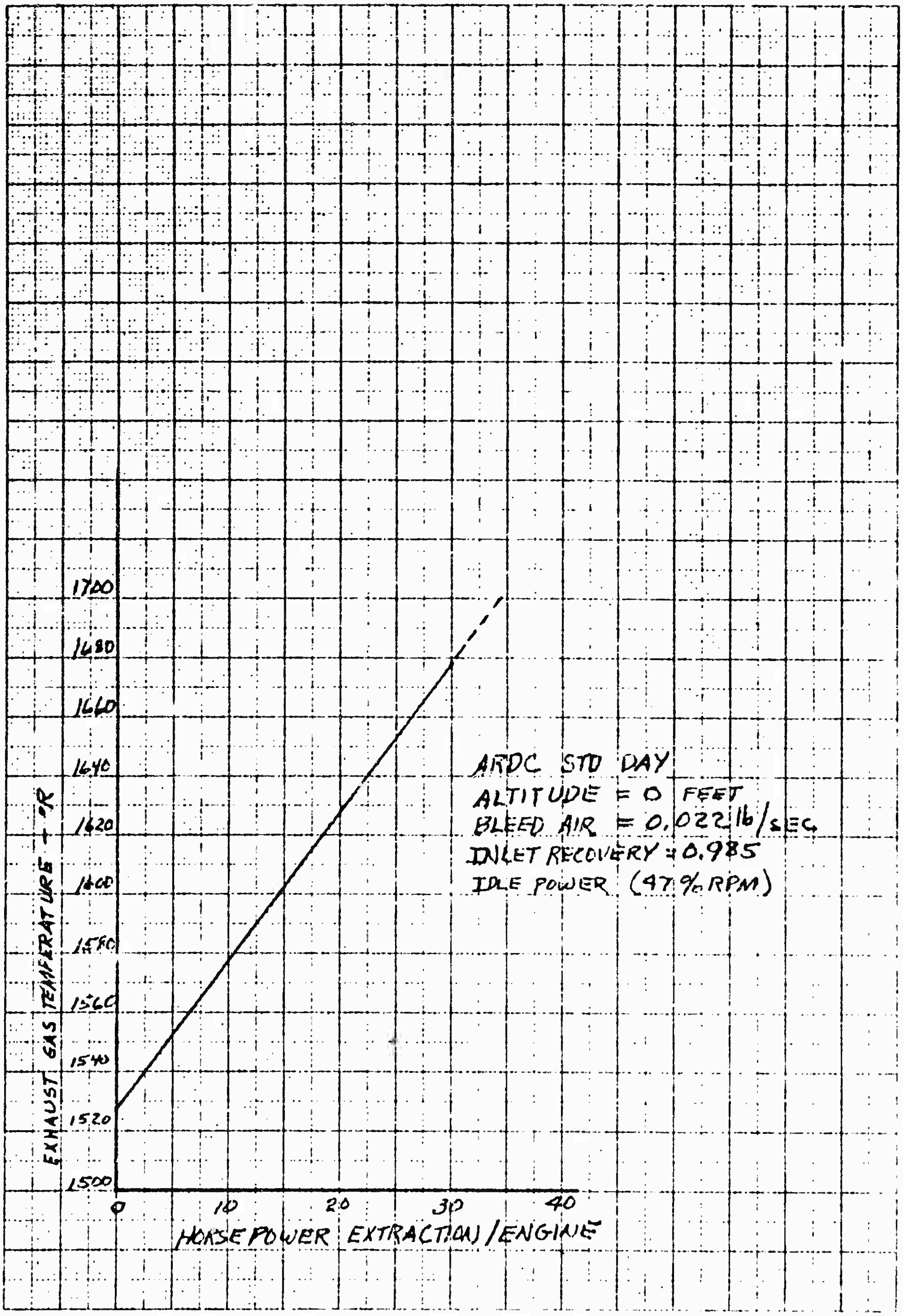

Figure 4.108 Effect of Horsepower Extraction on Exhaust Gas Temperature

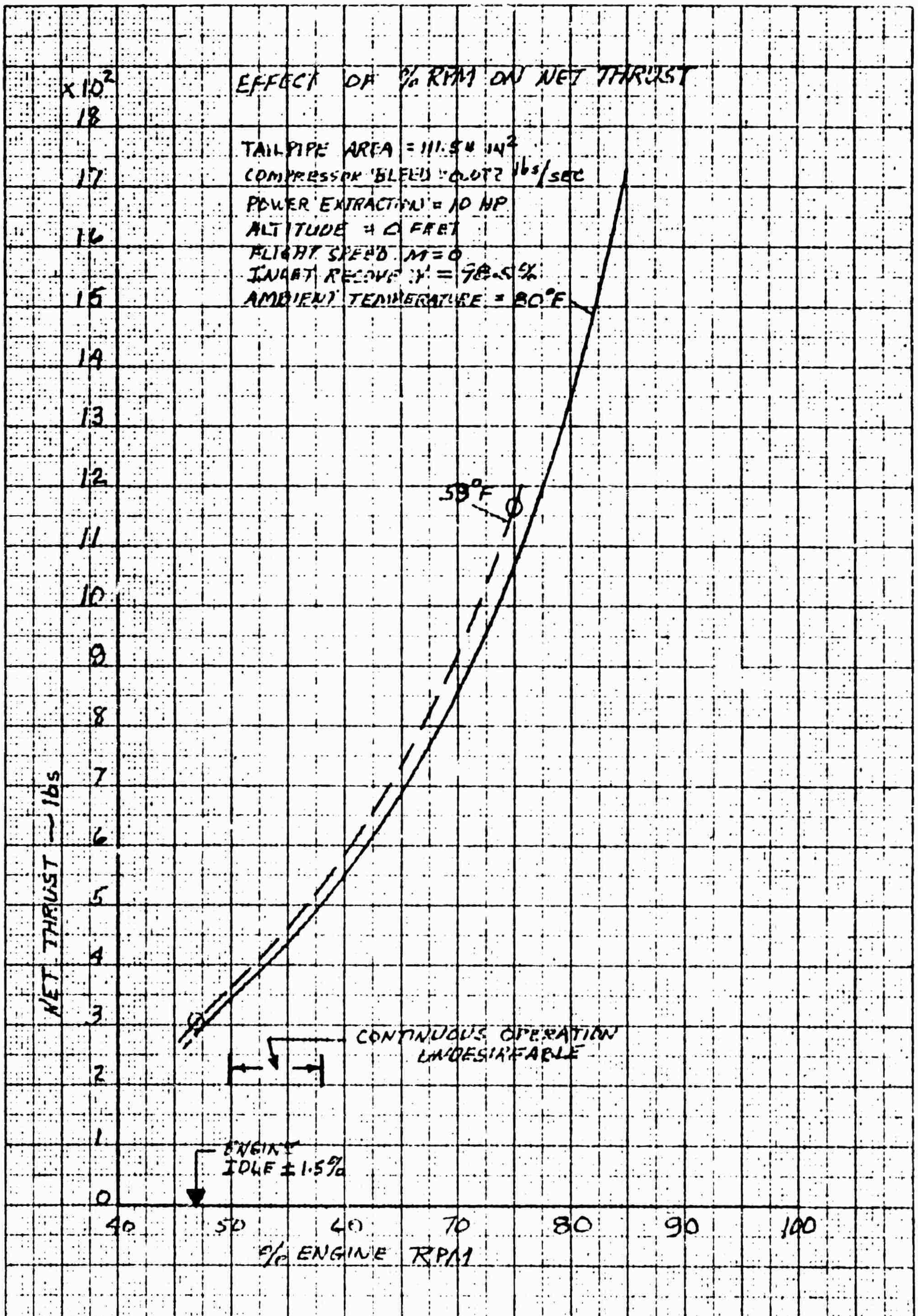

Figure 4.109 Effect of % RPM on Net Thrust; Idle Power to 85% RPM

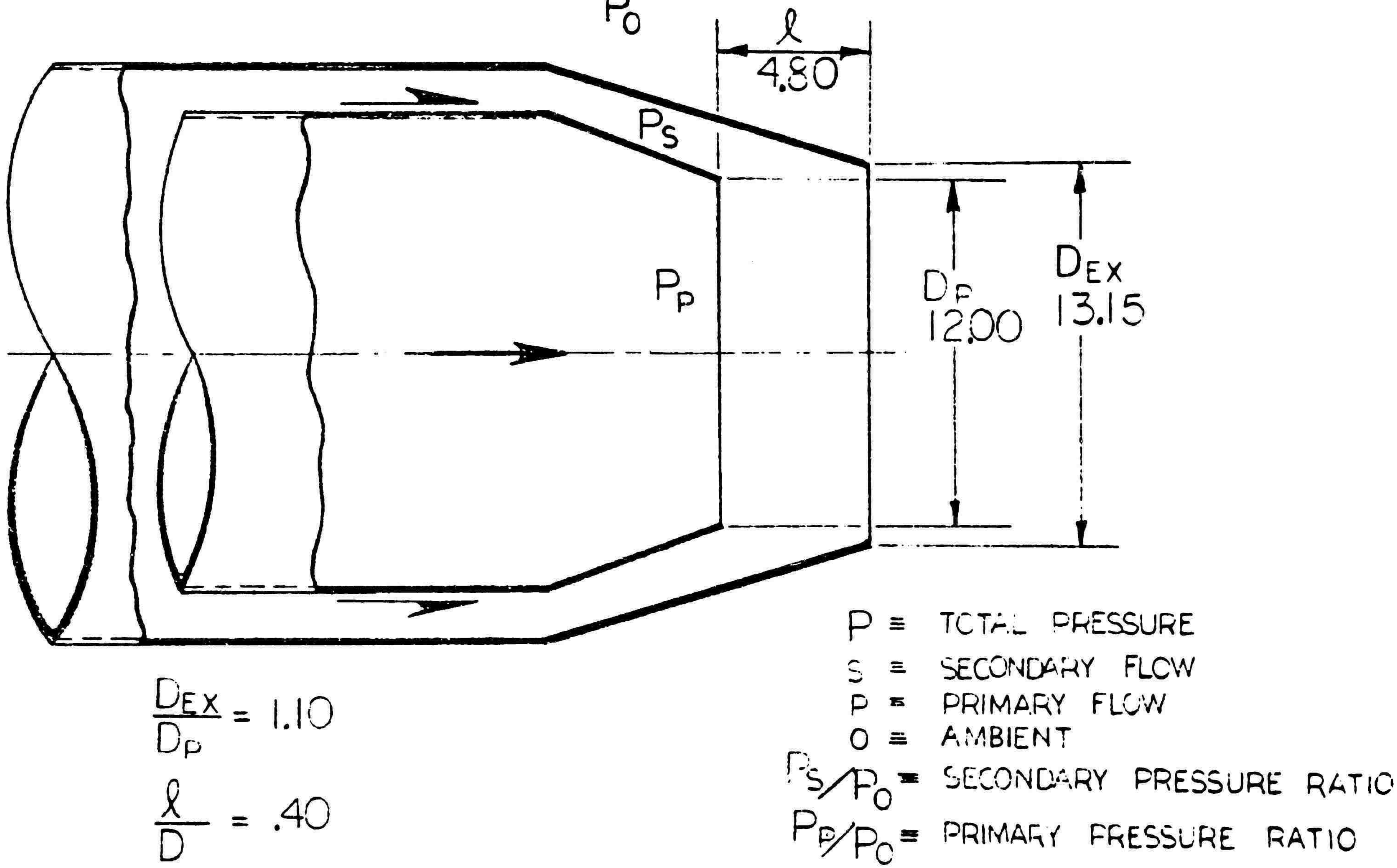

Figure 4.110 XV-5A Ejector Configuration

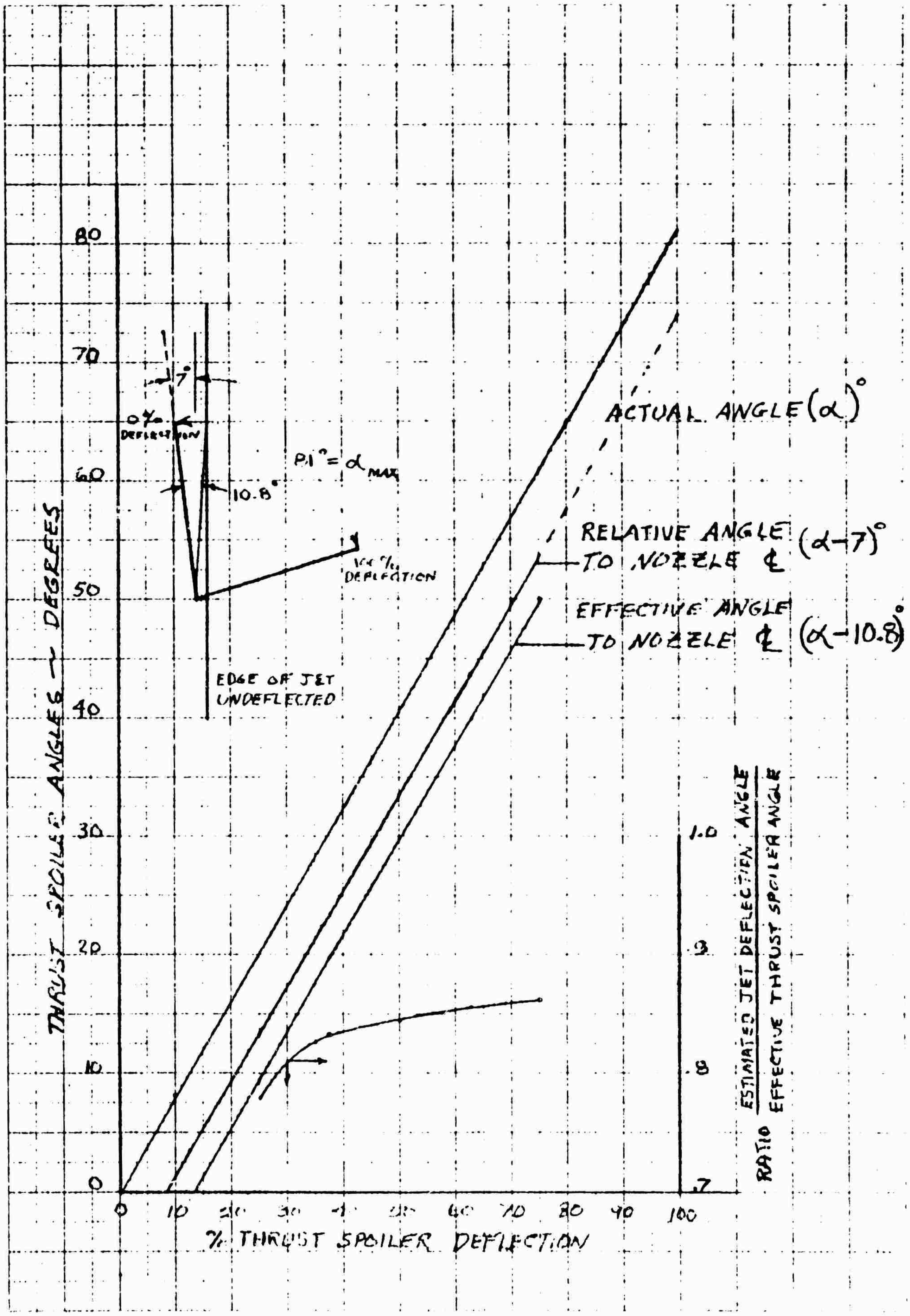

Figure 4.111 Thrust Spoiler Angles and Jet Deflection Effectiveness vs % Thrust Spoiler Deflection

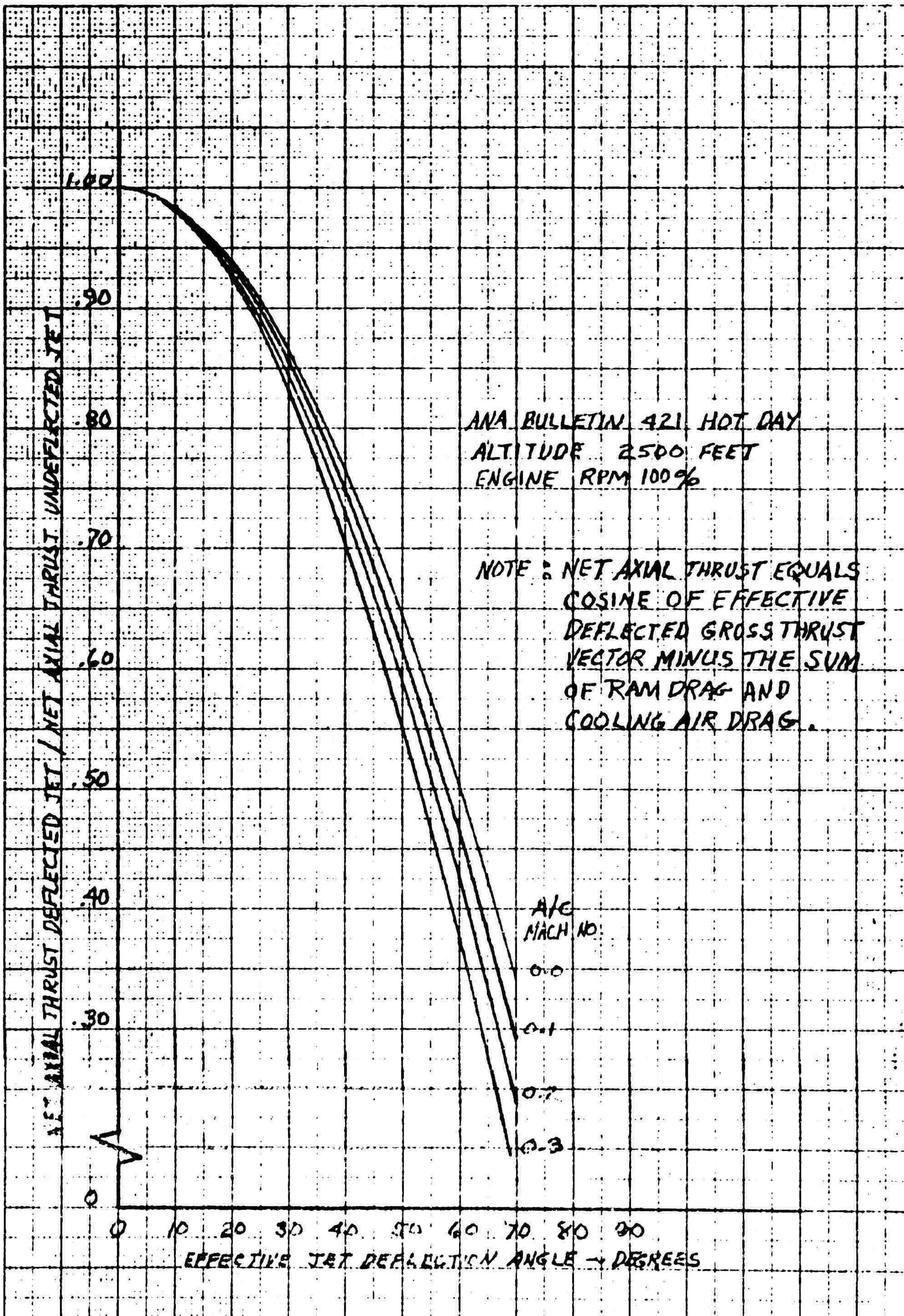

Figure 4.112 Reduction in Axial Net Thrust vs Effective Jet Deflection Angle, Hot Day, 2500 feet Altitude, 100% Engine RPM

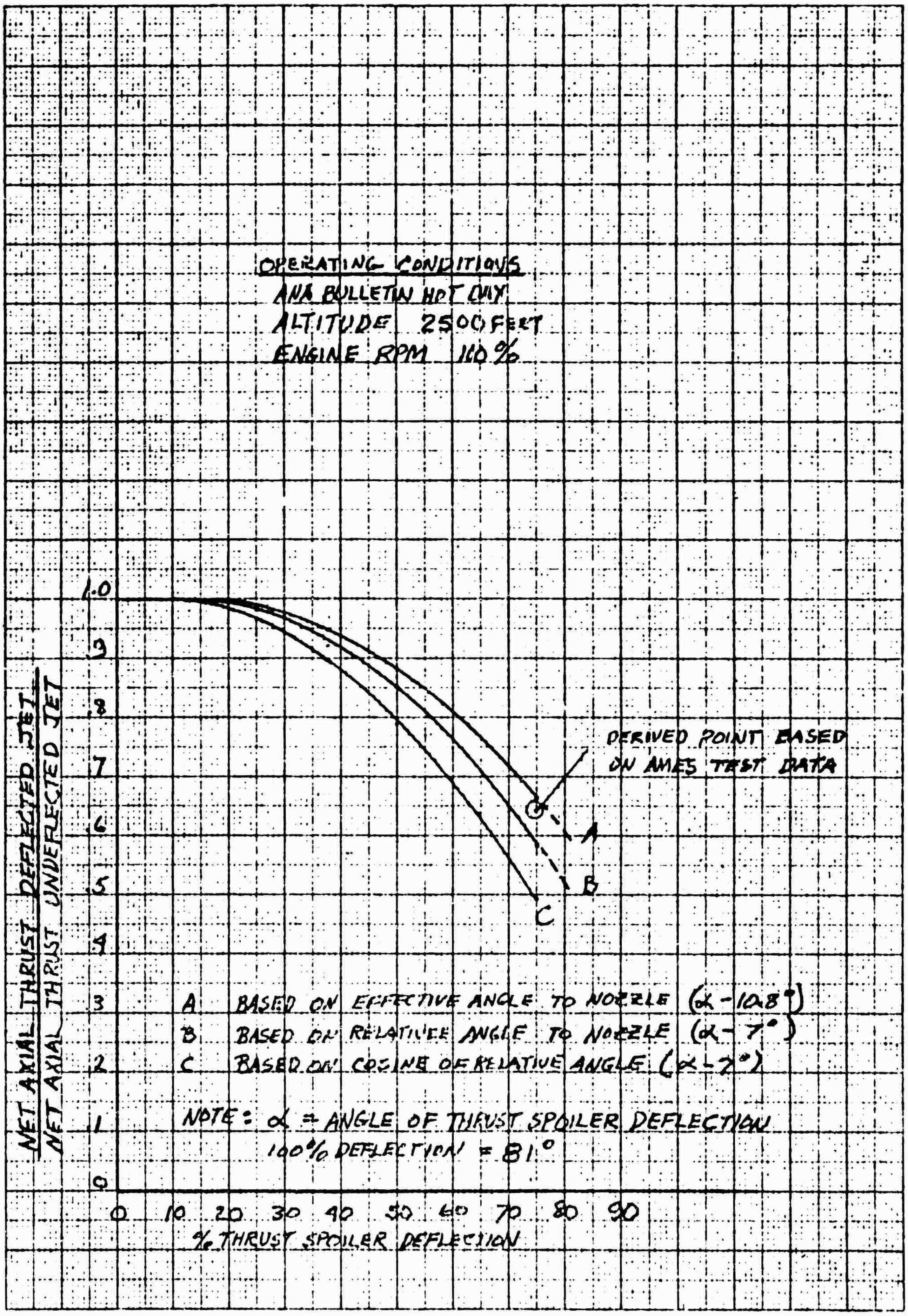

Figure 4.113 Reduction in Axial Net Thrust vs % Thrust Spoiler Deflection - Hot Day, 2500 feet Altitude, 100% RPM

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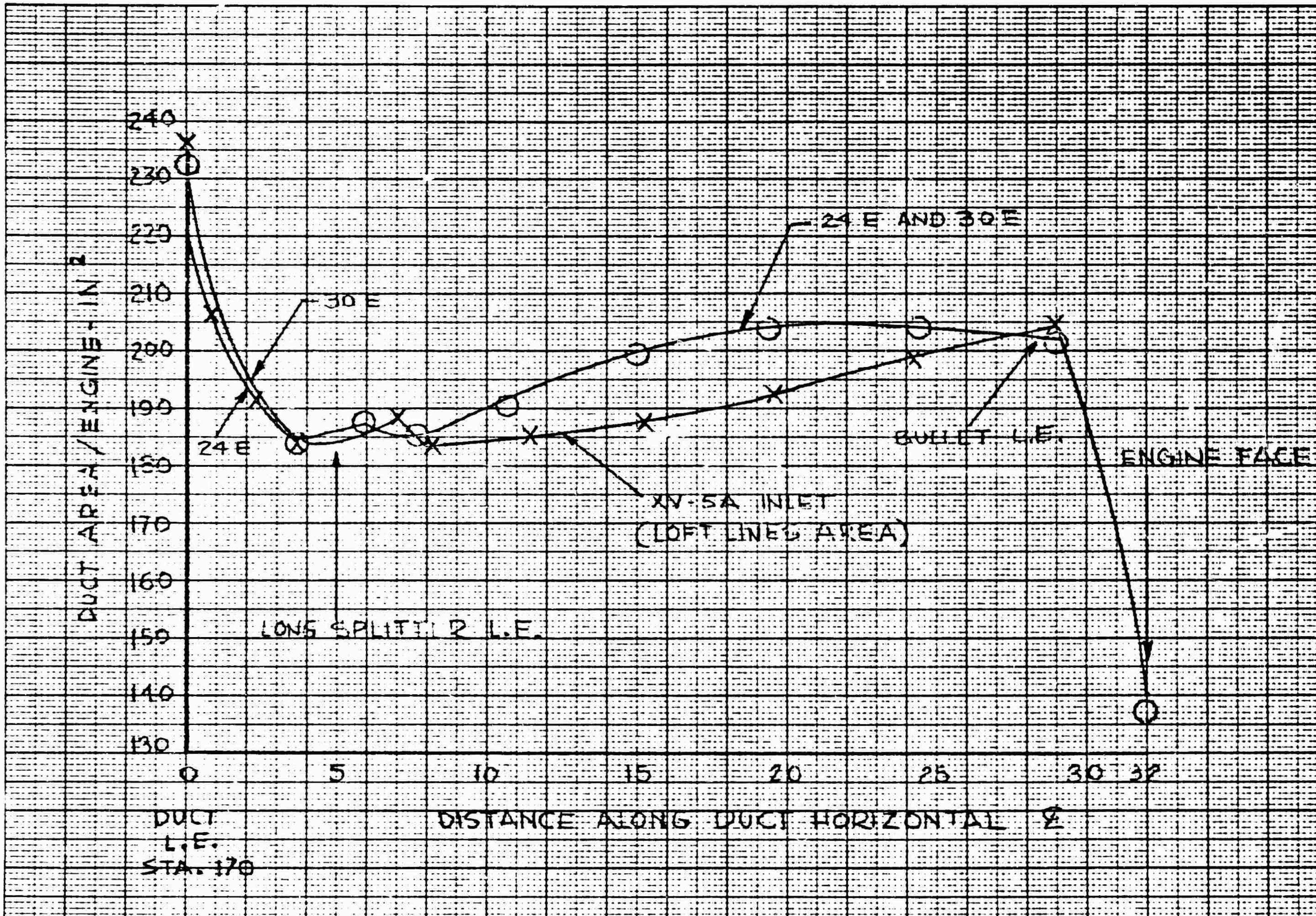

Figure 4.114 XV-5A Inlet Area Profile

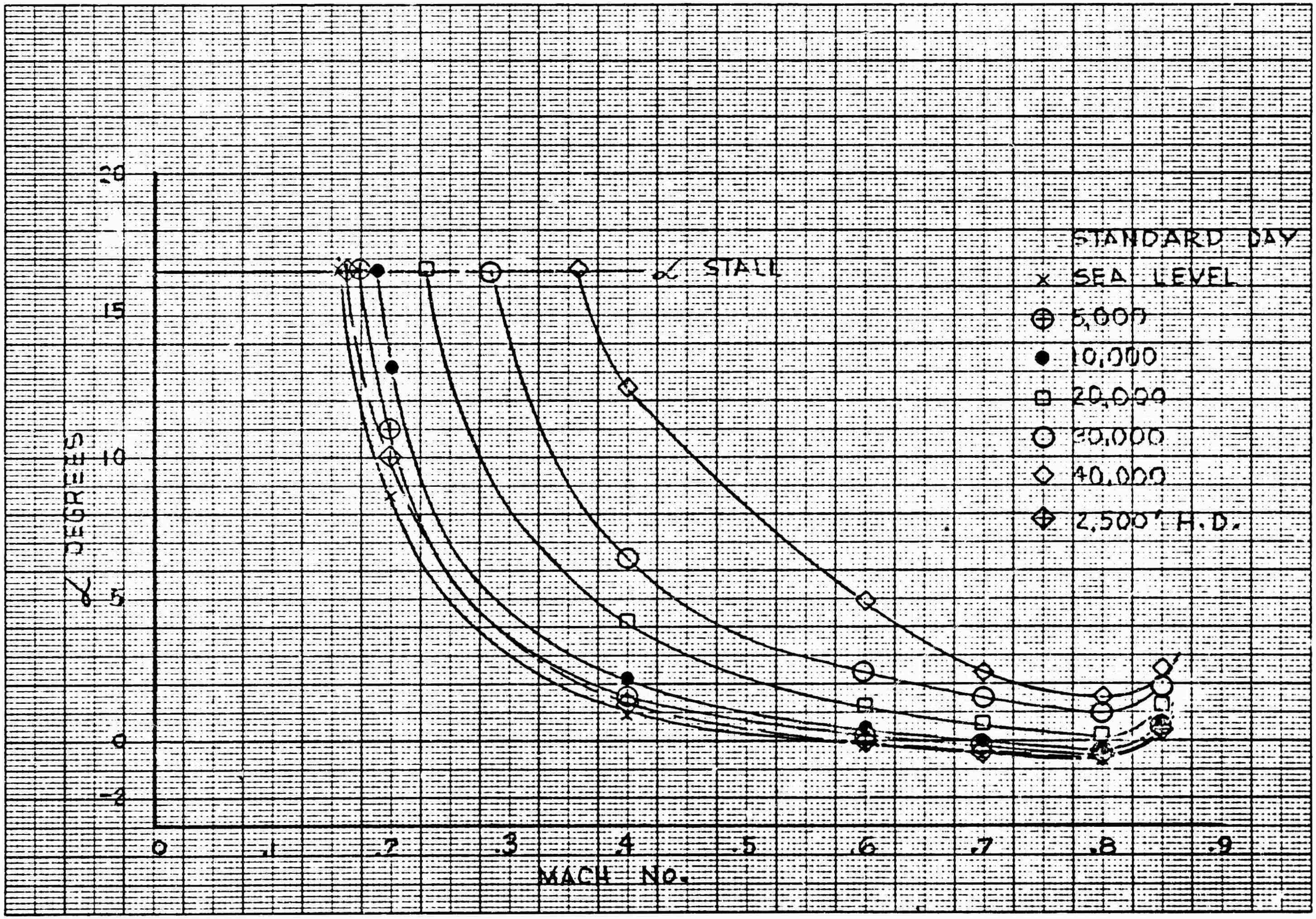

Figure 4.115 Typical XV-5A Trimmed Flight Attitude vs. Mach No. and Altitude

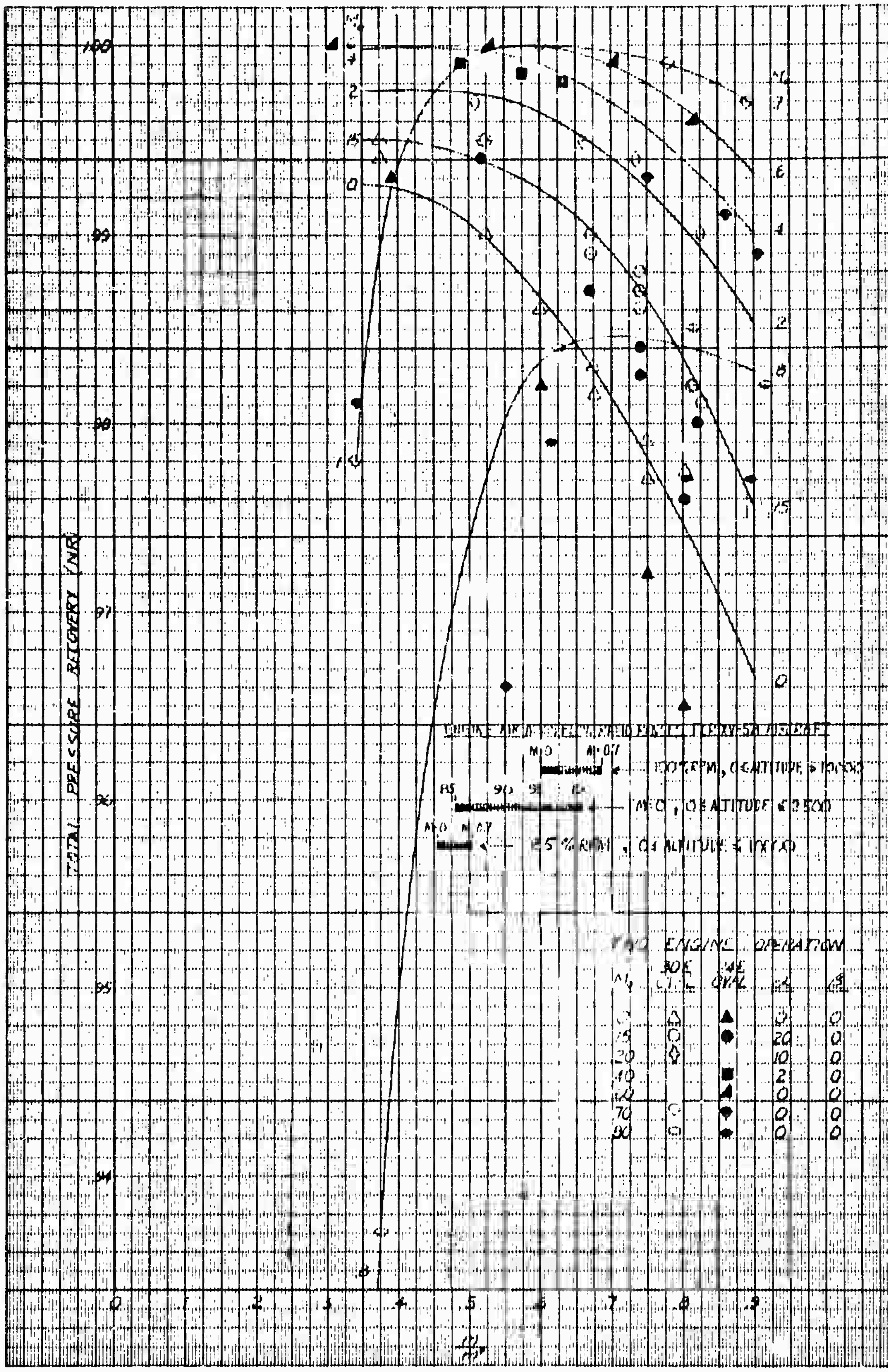

Figure 4.116 XV-5A Inlet Total Pressure Recovery vs Mass Flow Ratio and Mach No. for Two Engine Operation

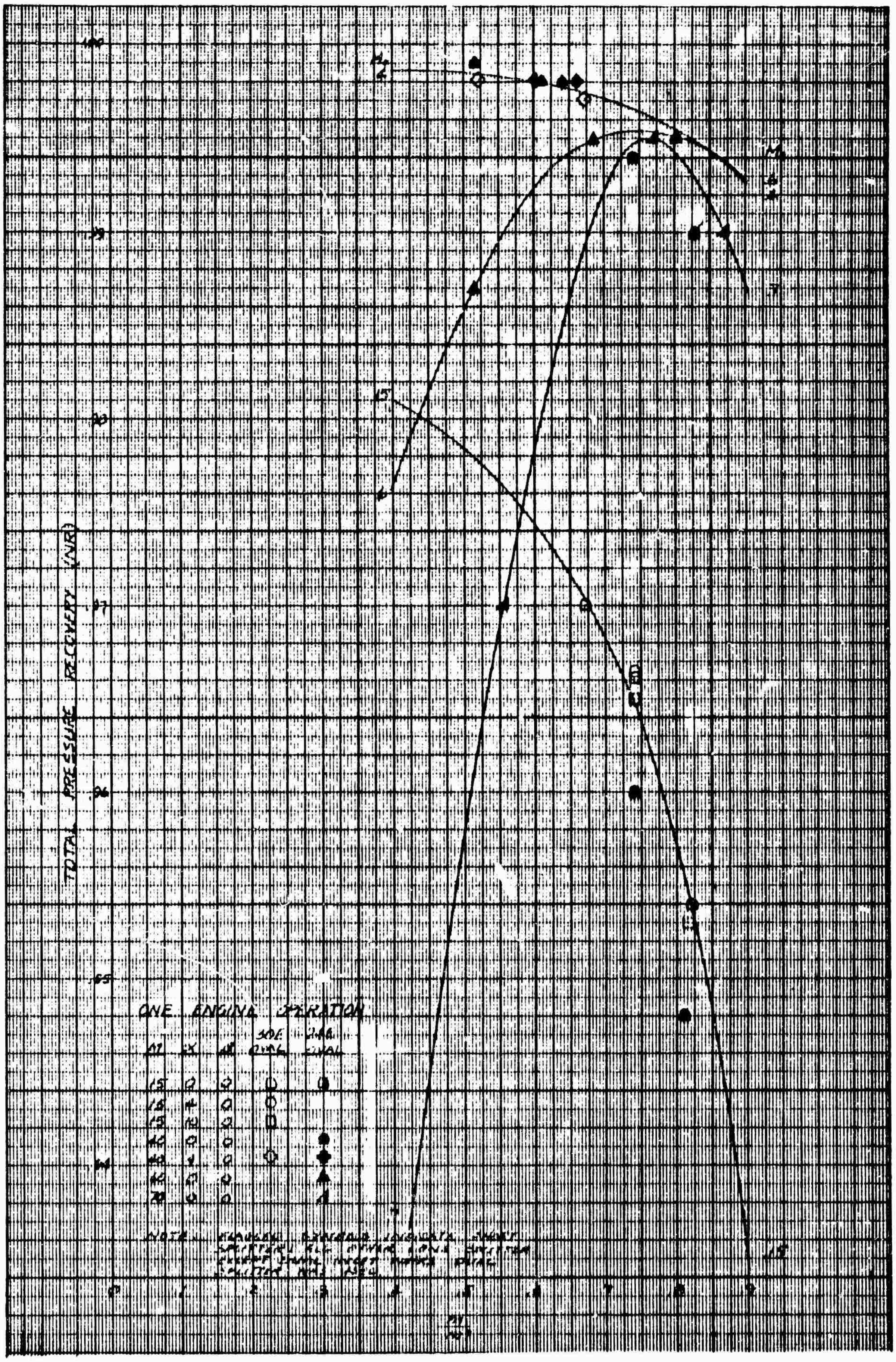

Figure 4.117 XV-5A Inlet Total Pressure Recovery vs Mass Flow Ratio and Mach No. for Single Engine Operation

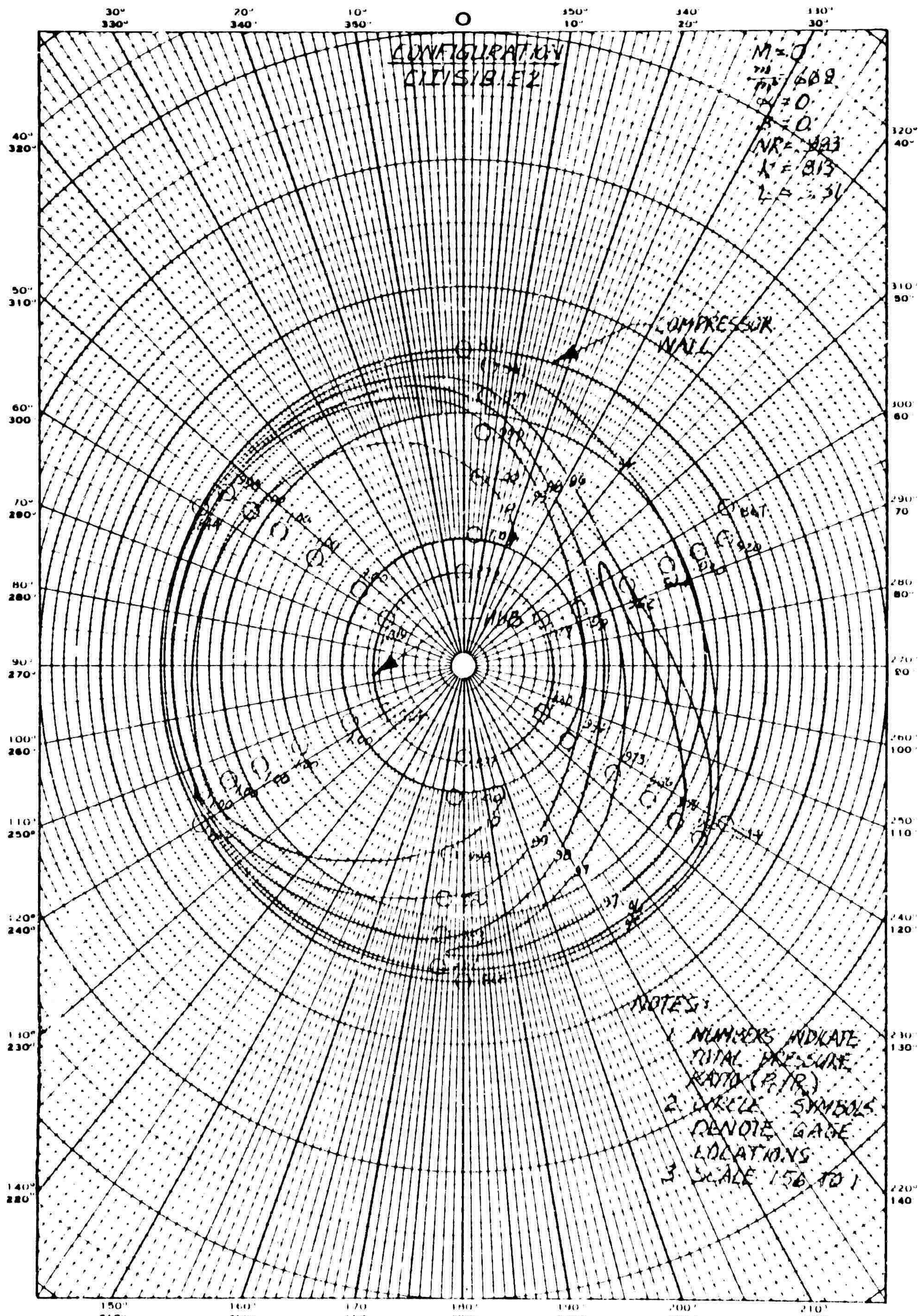

Figure 4.118 Total Pres[illegible]e Distribution at Left/Hand Engine Compressor Face; M = 0, α = [illegible] = 0, 30E Oval Inlet, 2 Engine Operation

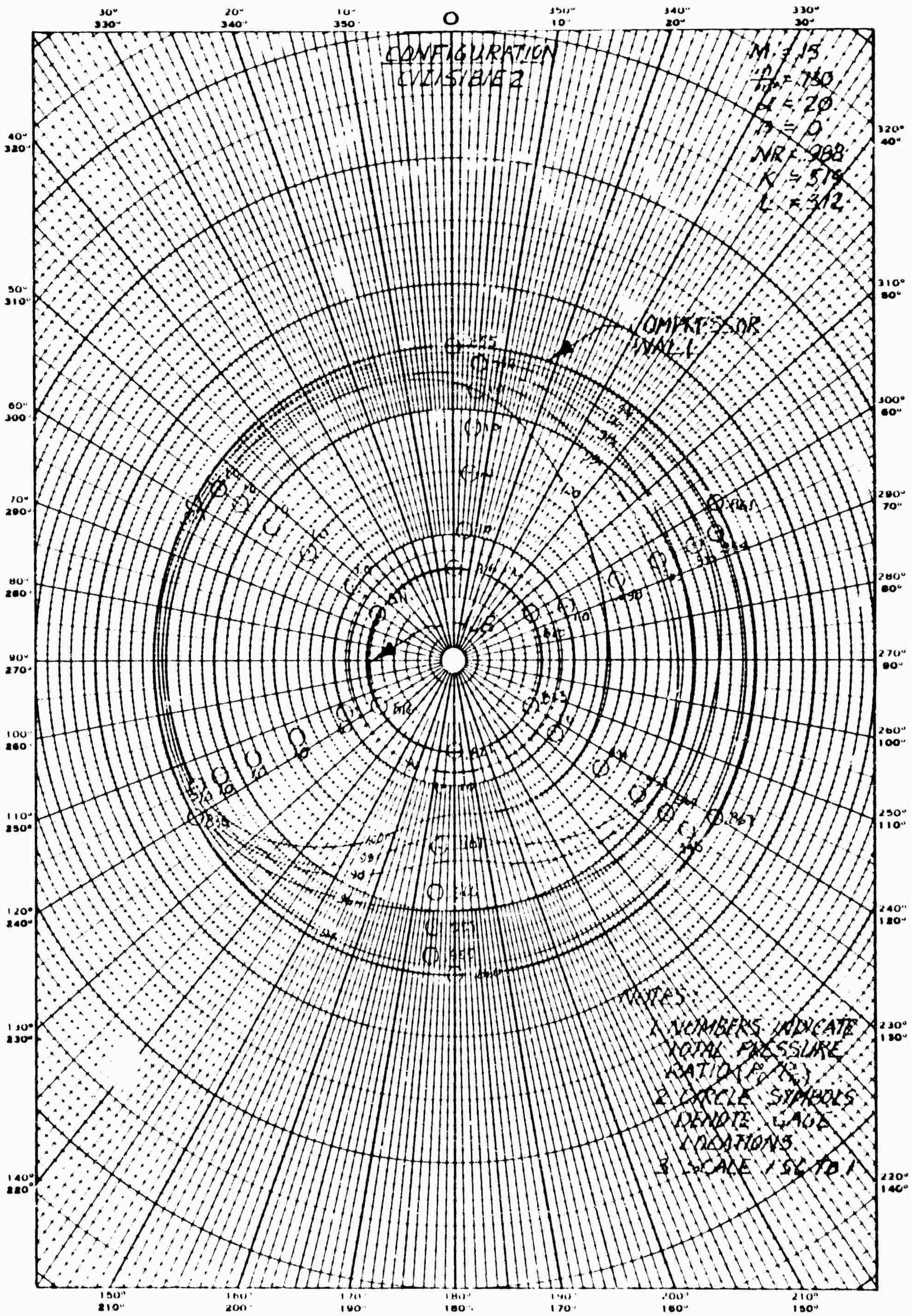

Figure 4.119 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .15, $\alpha = 20$, $\beta = 0$, 30E Oval Inlet, 2 Engine Operation

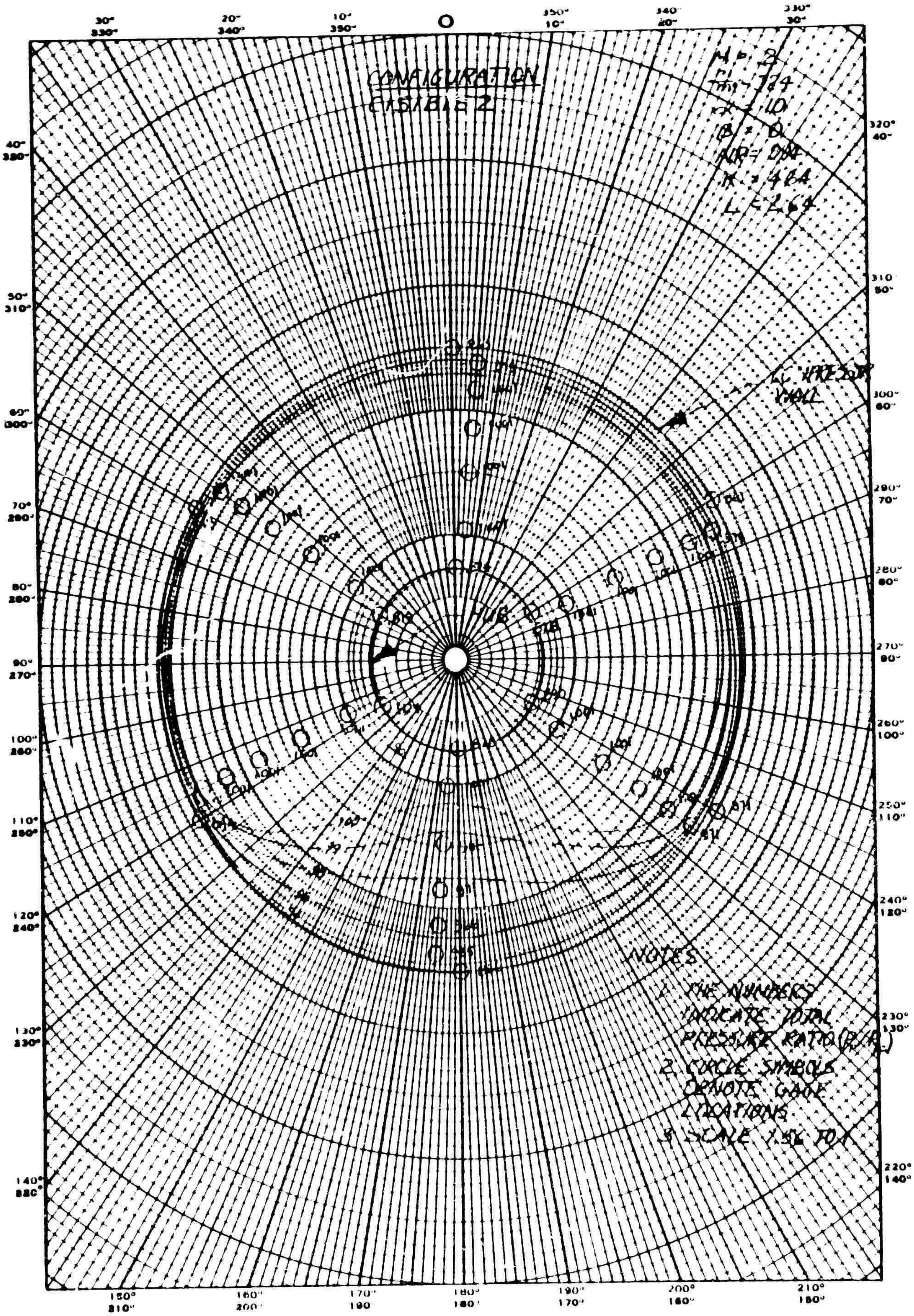

Figure 4.120 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .2, α =10, β = 0, 30E Oval Inlet, 2 Engine Operation

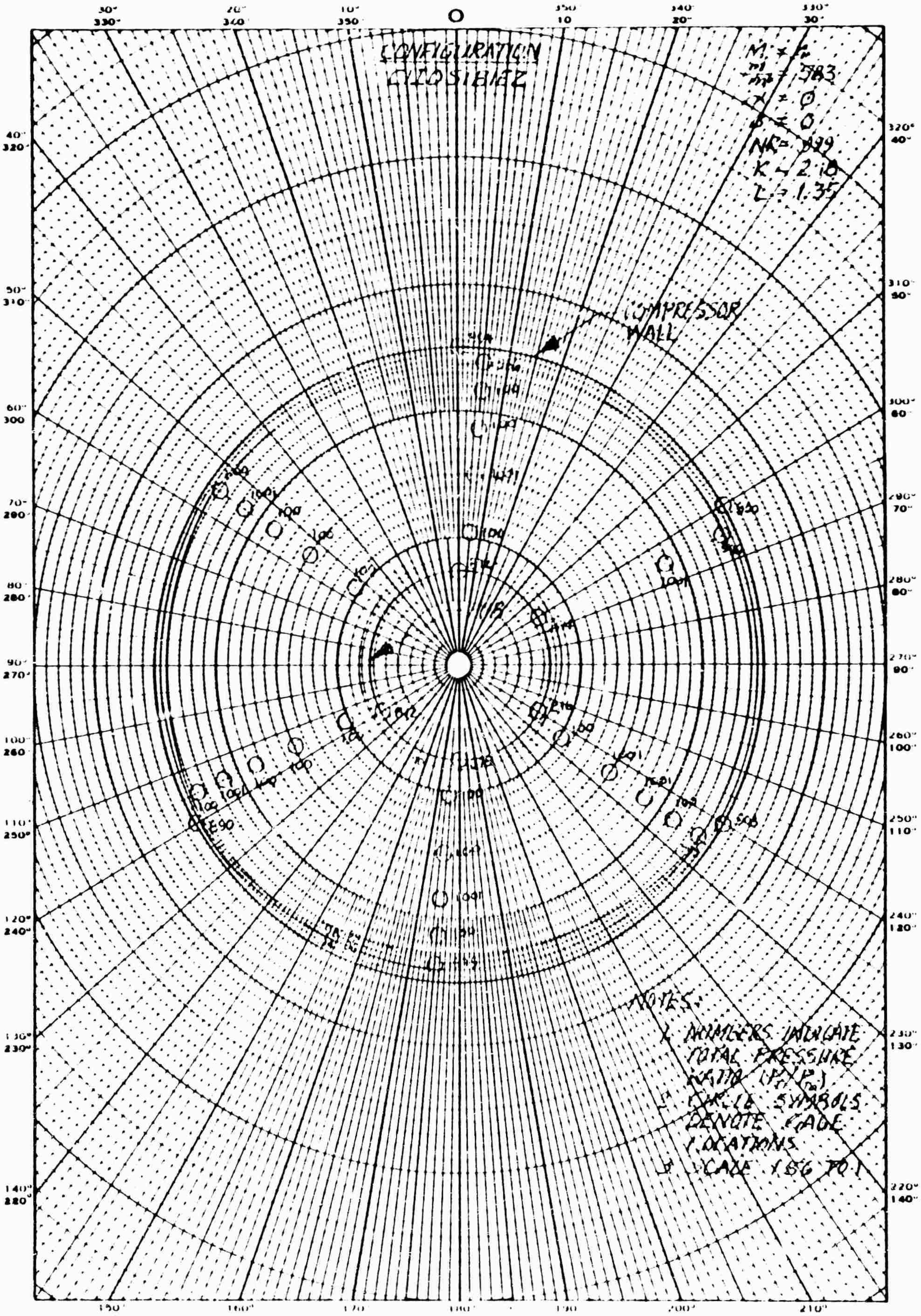

Figure 4.121 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .4, $\alpha = 0$, $\beta = 0$, 24E Oval Inlet, 2 Engine Operation

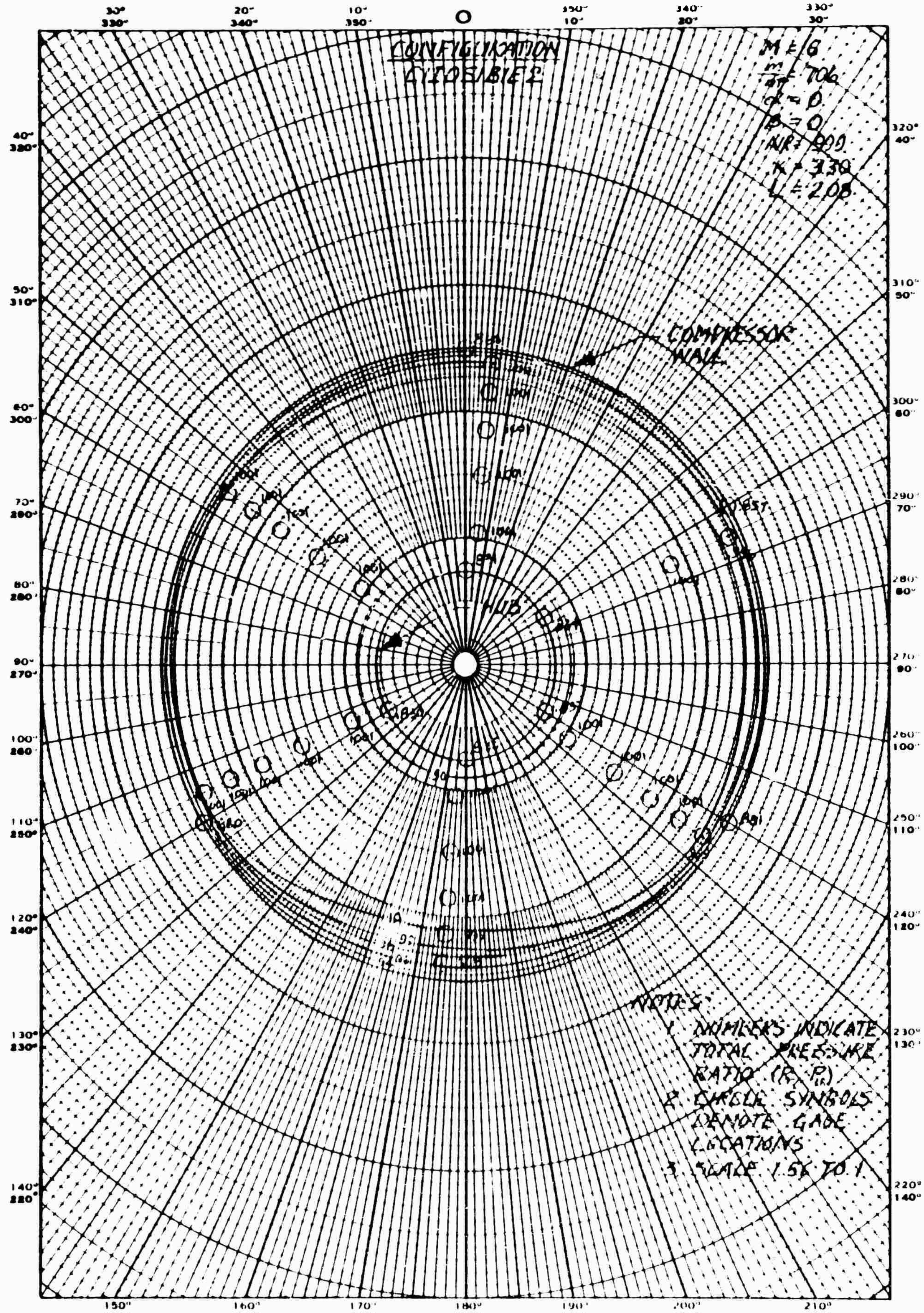

Figure 4.122 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .6, $\alpha = 0$, $\beta = 0$, 24E Oval Inlet, 2 Engine Operation

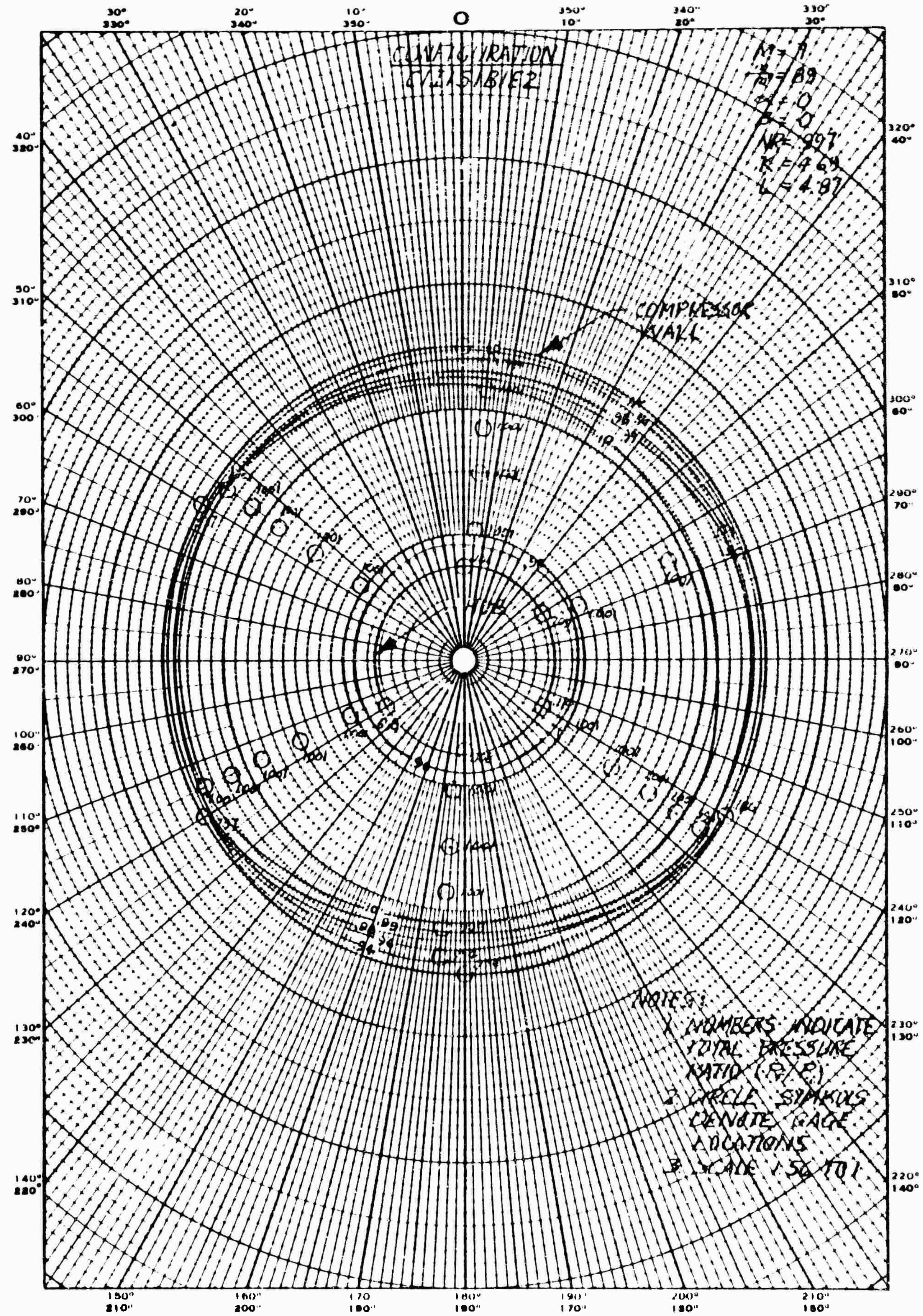

Figure 4.123 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .7, α = 0, β = 0, 30E Oval Inlet, 2 Engine Operation

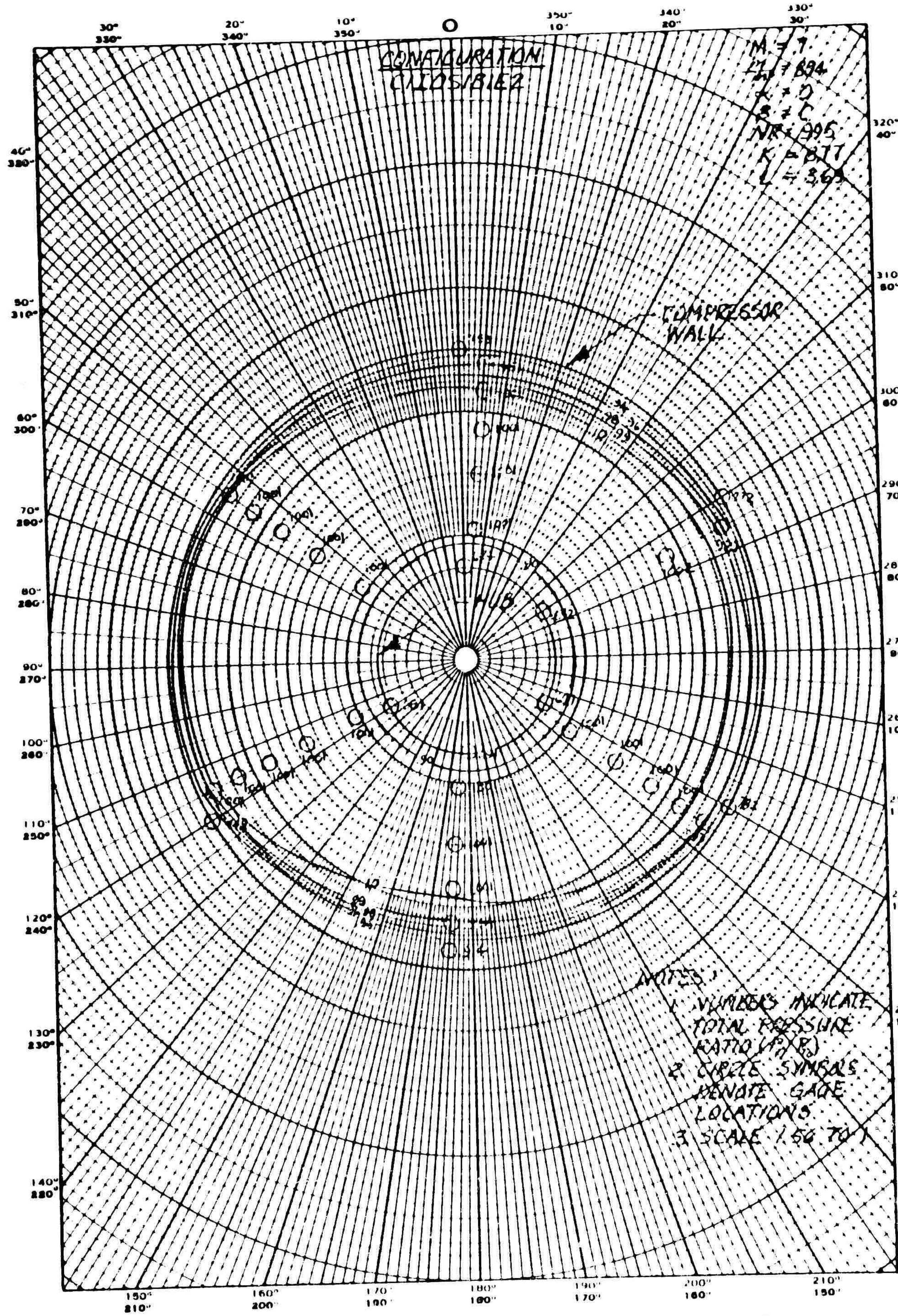

Figure 4.124 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .7, $\alpha = 0$, $\beta = 0$, 24E Oval Inlet, 2 Engine Operation

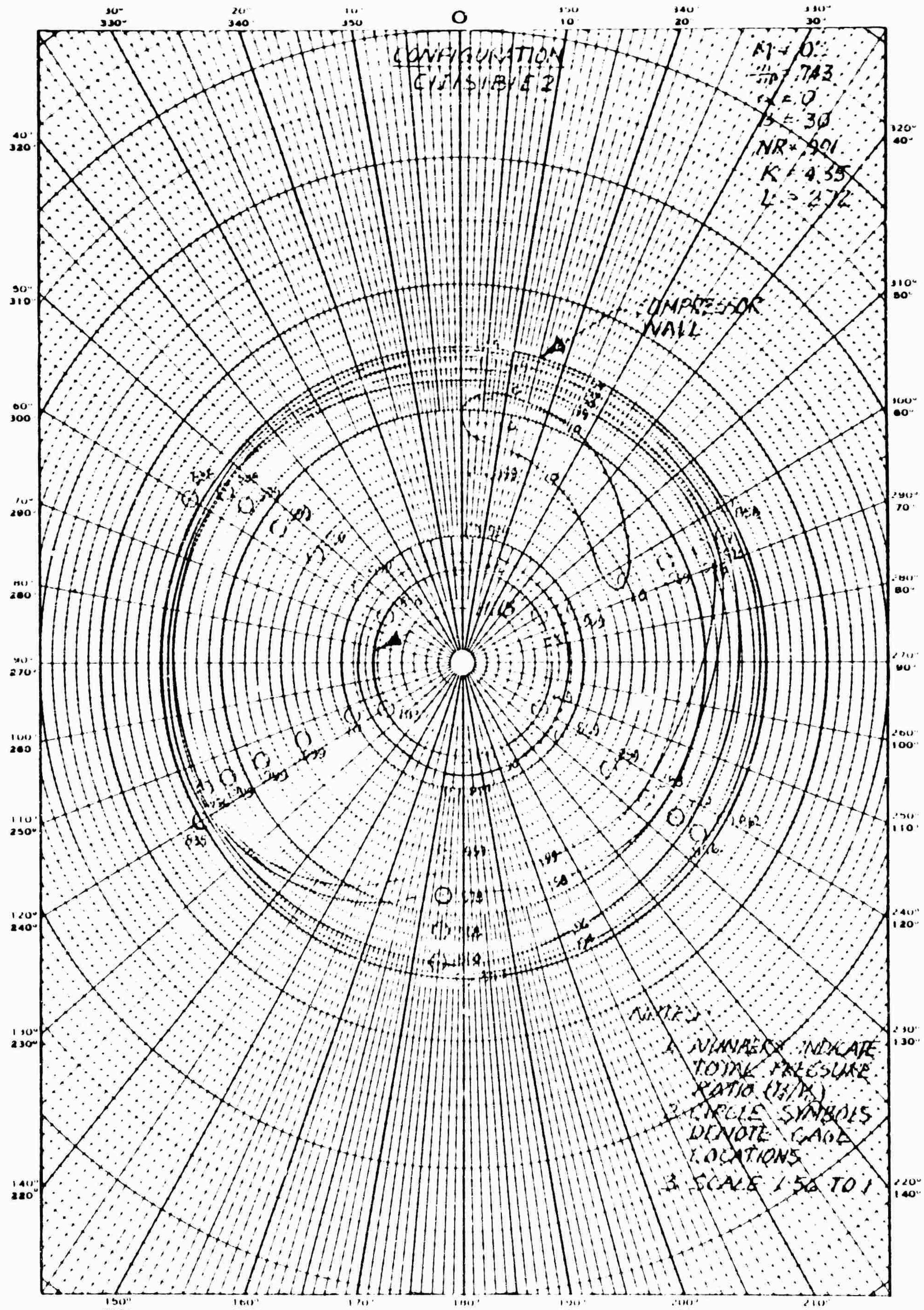

Figure 4.125 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .02, $\alpha = 0$, $\beta = 30$, M/M* = .743, 30E Oval Inlet, 2 Engine Operation

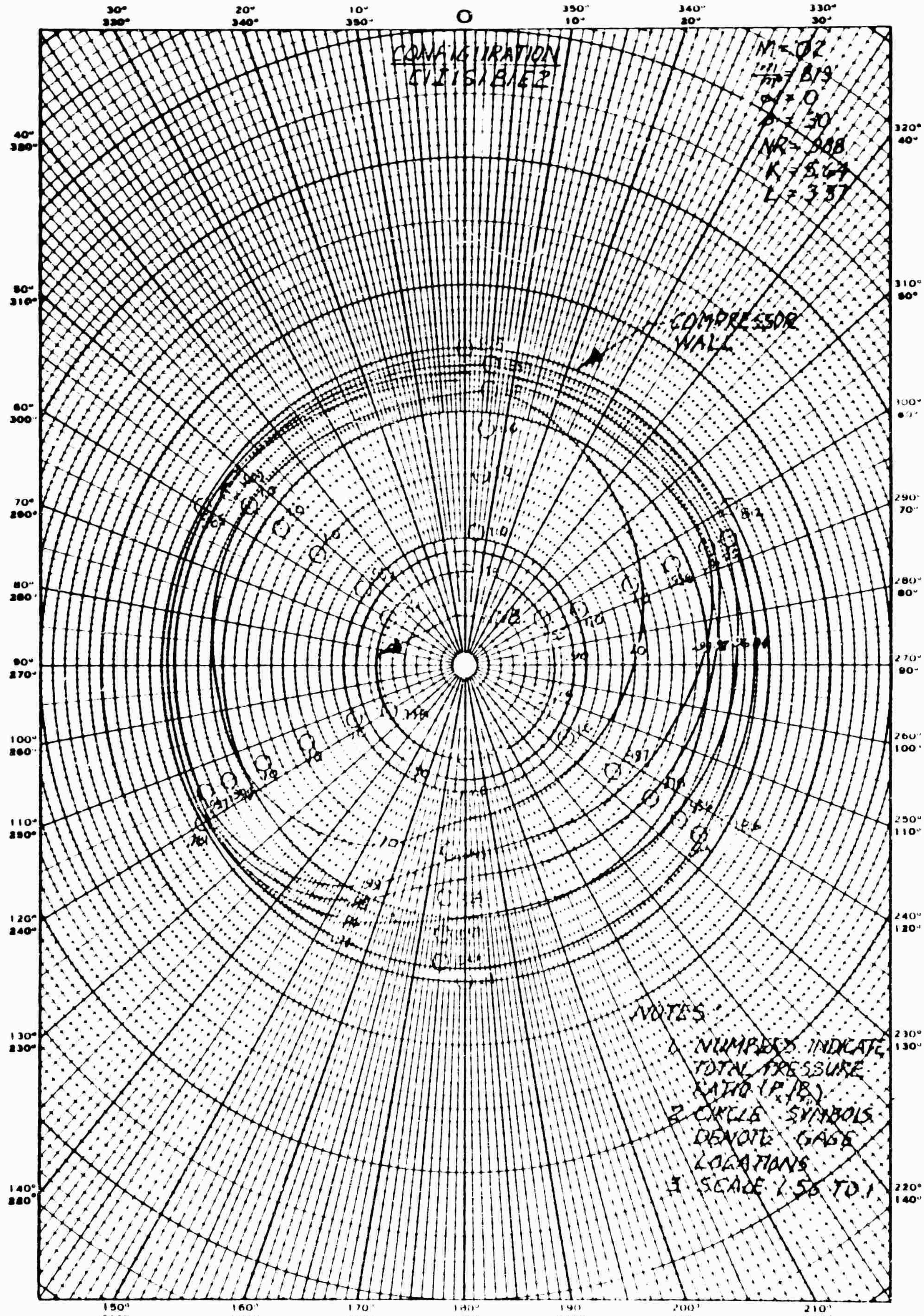

Figure 4.126 Total Pressure Distribution at Left Hand Engine Compressor Face; M = .02, $\alpha = 0$, $\beta = 30$, M/M* = .819, 30E Inlet, 2 Engine Operation

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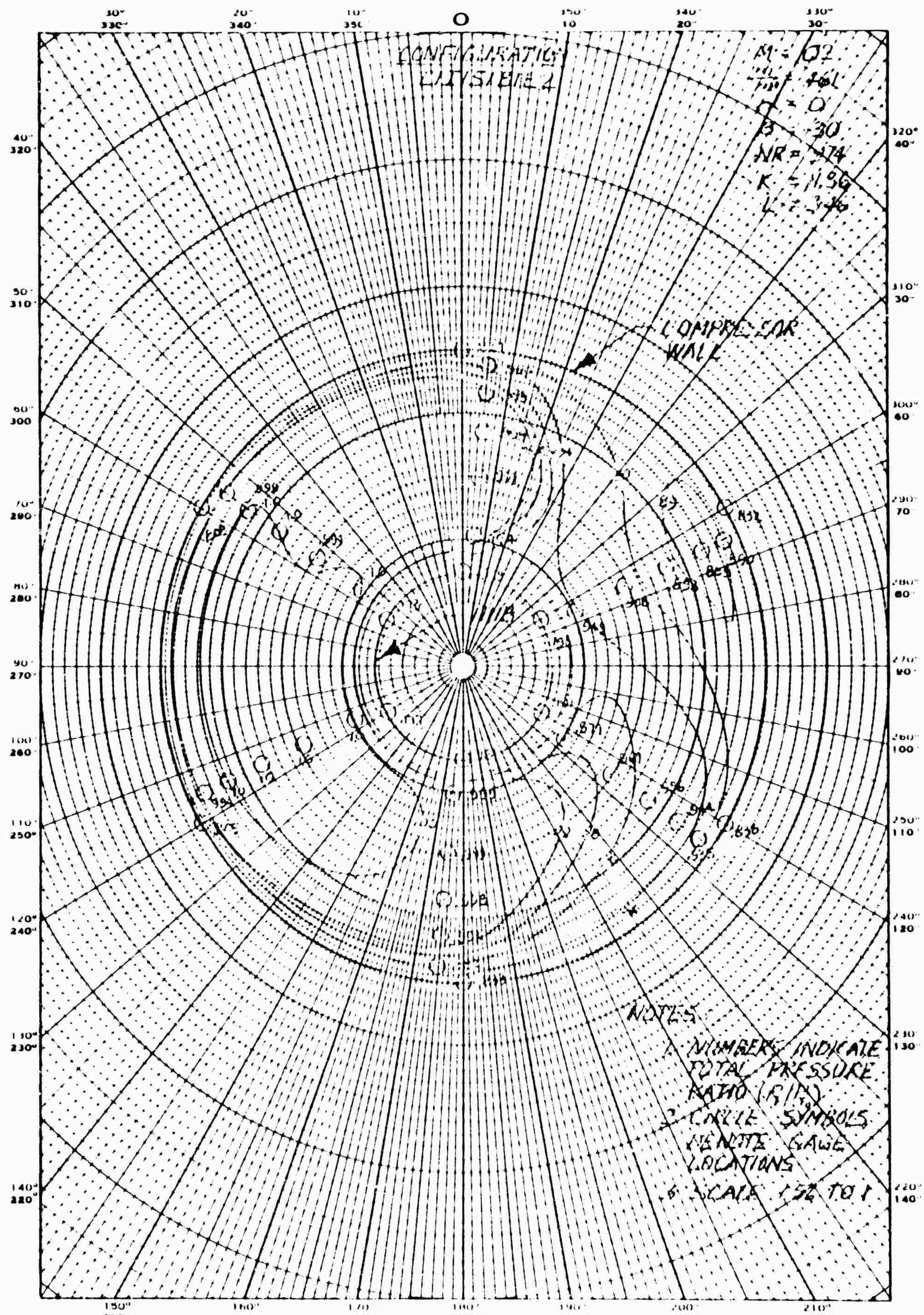

Figure 4.127 Total Pressure Distribution at Left Hand Engine Compressor Face; M=.02, α=0, β=-30, M/M*=.461, 30E Inlet, 2 Engine Operation

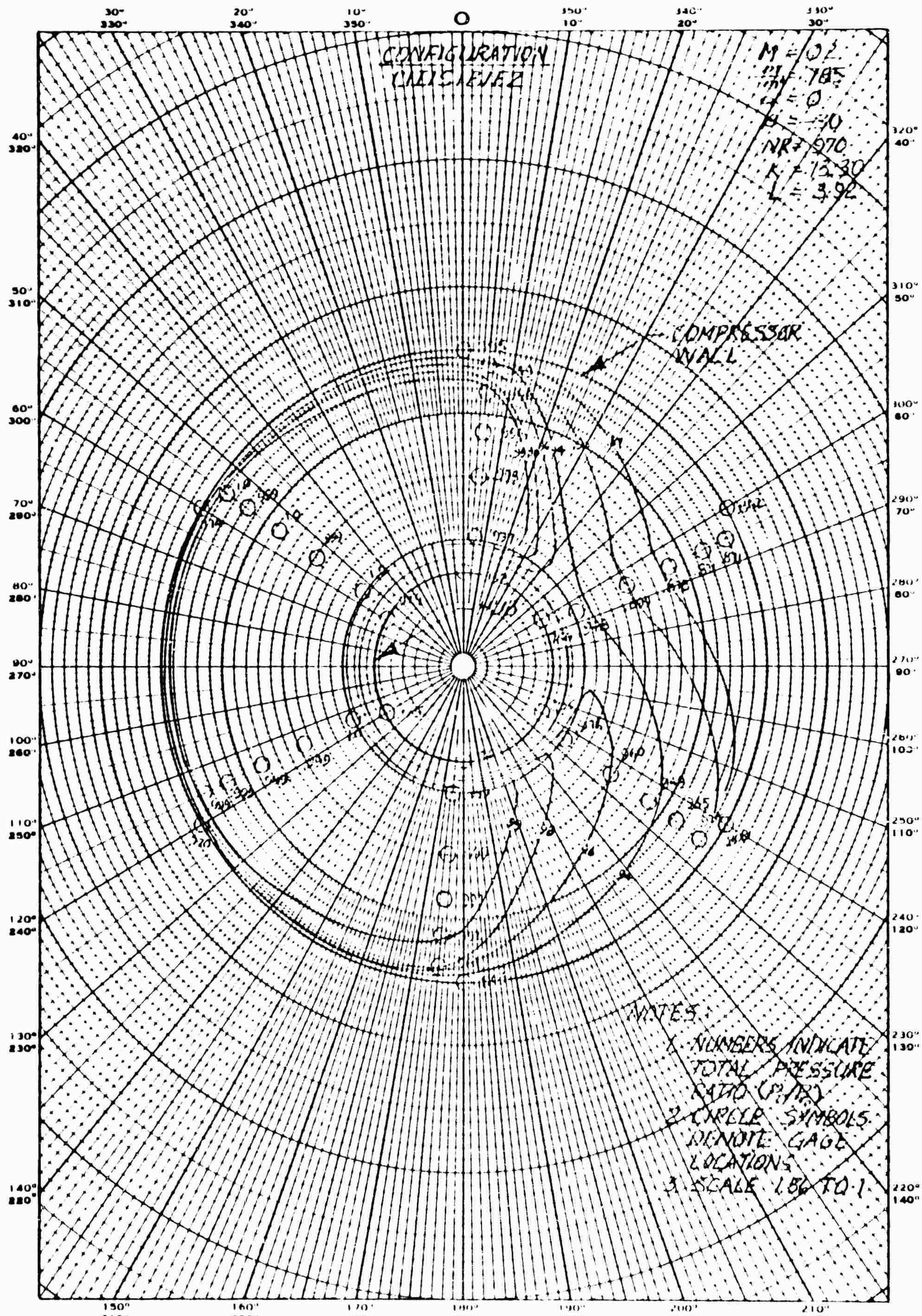

Figure 4.128 Total Pressure Distribution at Left Hand Engine Compressor Face; M=.02 α=0, β=-30, M/M*=.785, 30E Inlet, 2 Engine Operation

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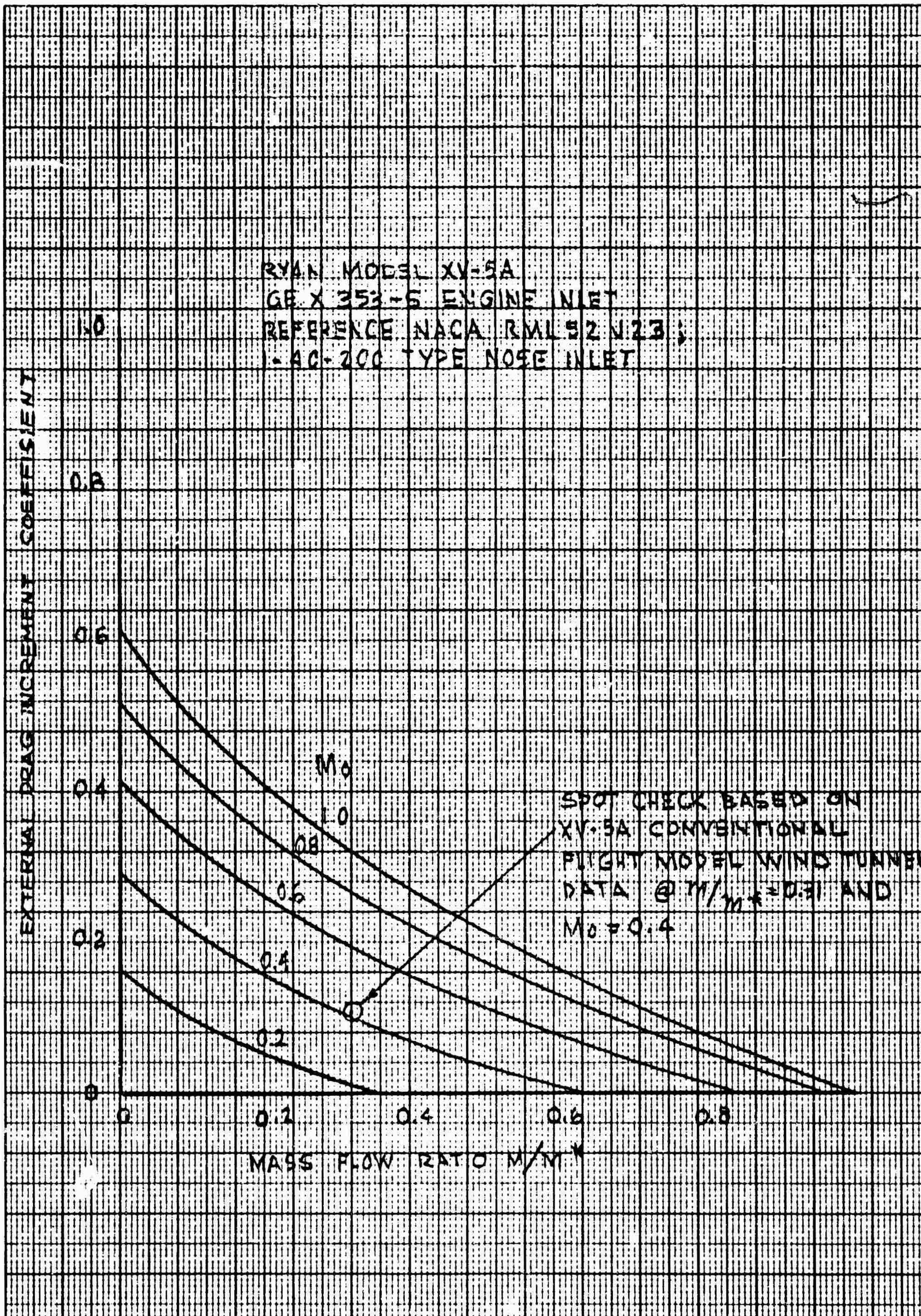

Figure 4.129 Estimated External Drag Increment Coefficient vs Mass Flow Ratio and Mach No.

5.0 XV-5A AIRCRAFT COOLING AIR DRAG

As pointed out in Section 4.0, cooling air drag must be subtracted from the net thrust in order to obtain thrust available for estimating XV-5A aircraft aerodynamic performance. These data are presented in Figures 5.1 through 5.26. The cooling air drag is determined in the same manner as engine air ram drag, knowing the total cooling system air taken on board in terms of the various aircraft and gas generator operating parameters. Total cooling air taken on board during turbojet mode conventional flight operation includes air entering the cockpit, the cooling-fan-plenum fuselage inlet, the boundary-layer bleed duct, and the pitch fan compartment. Recirculation of air through the wing fan cavities was found to be negligible for purposes of calculating cooling air drag. Details of the cooling system performance are presented in Reference 14.

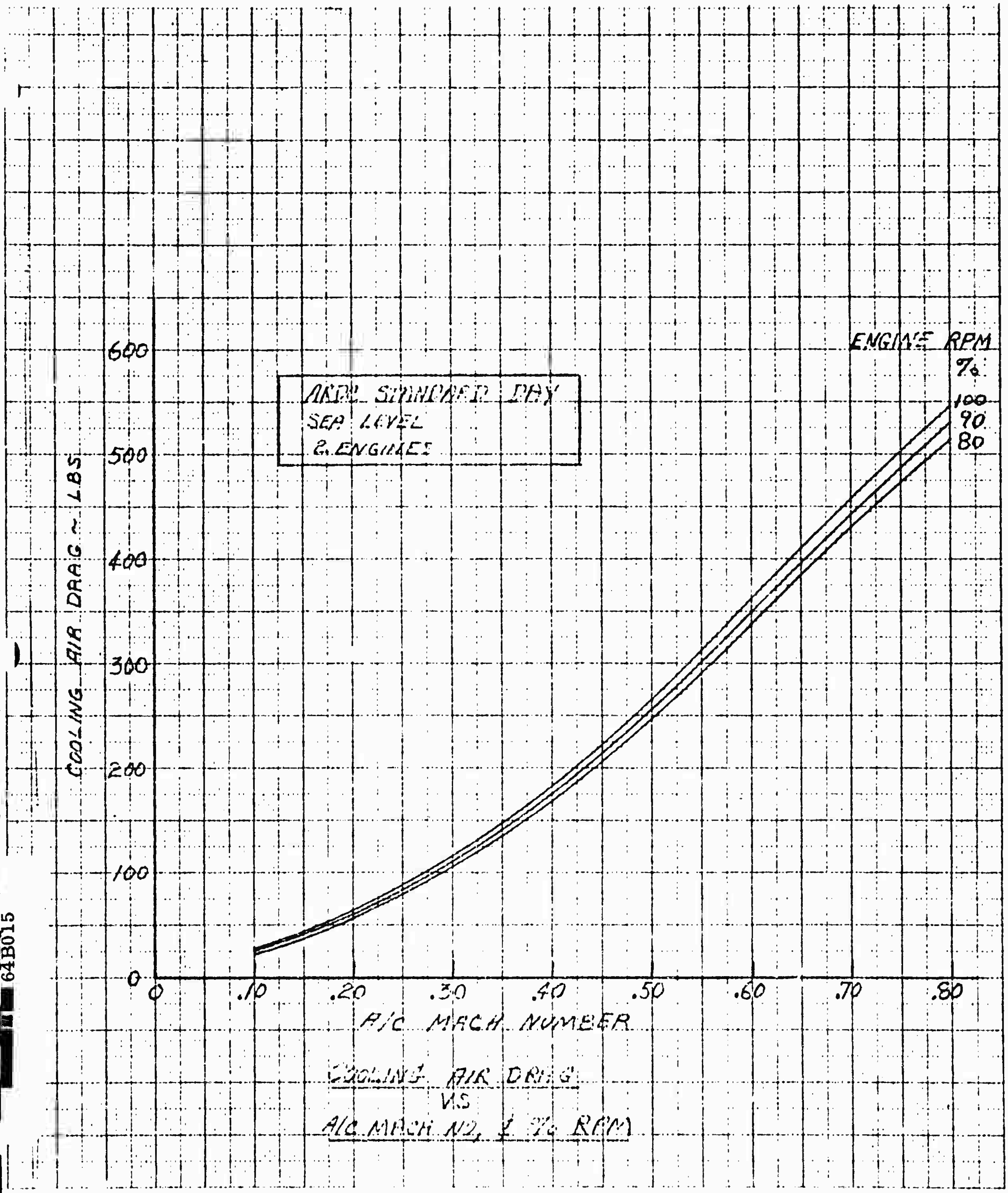

Figure 5.1 Cooling Air Drag vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Std. Day

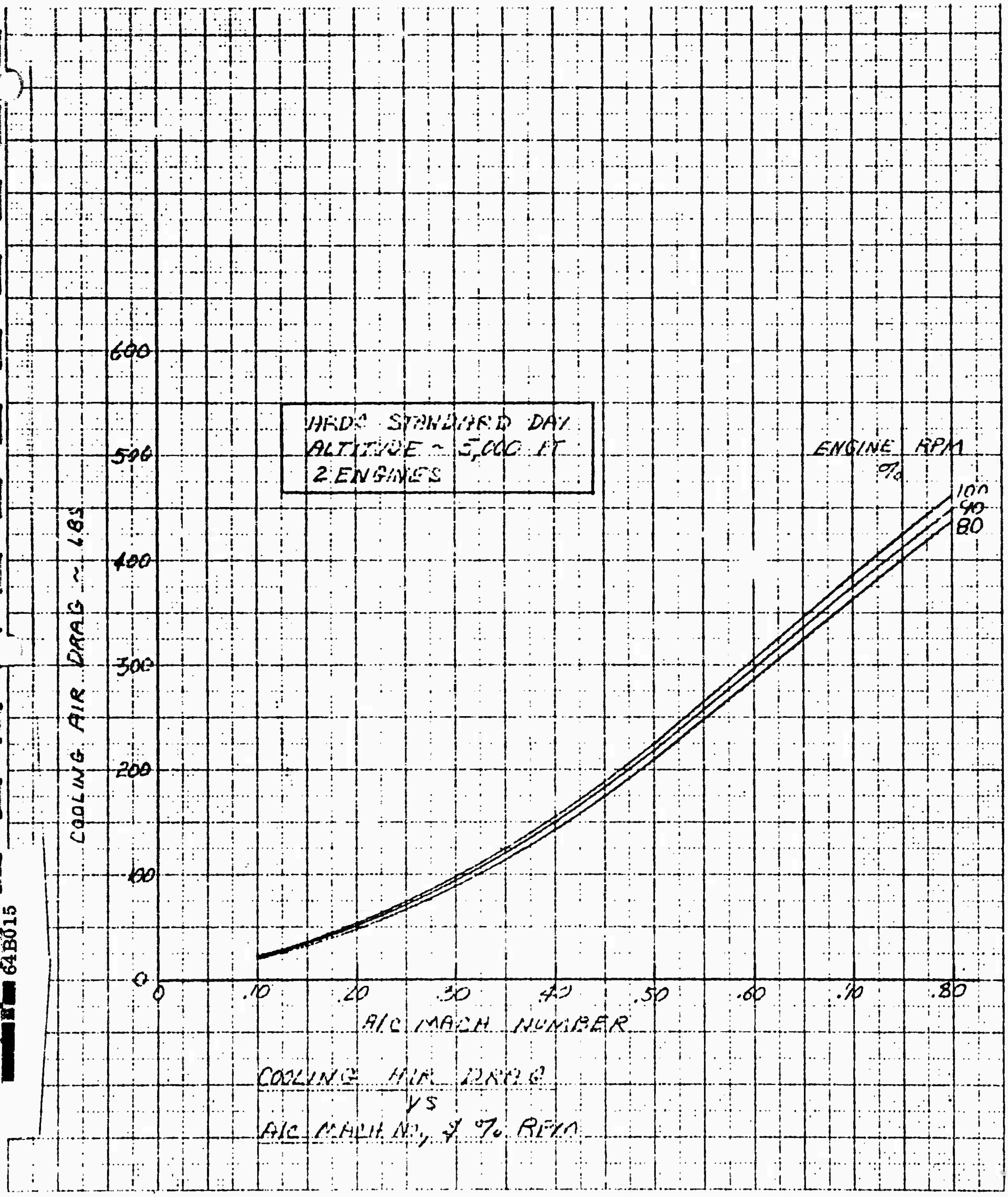

Figure 5.2 Cooling Air Drag vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Std. Day

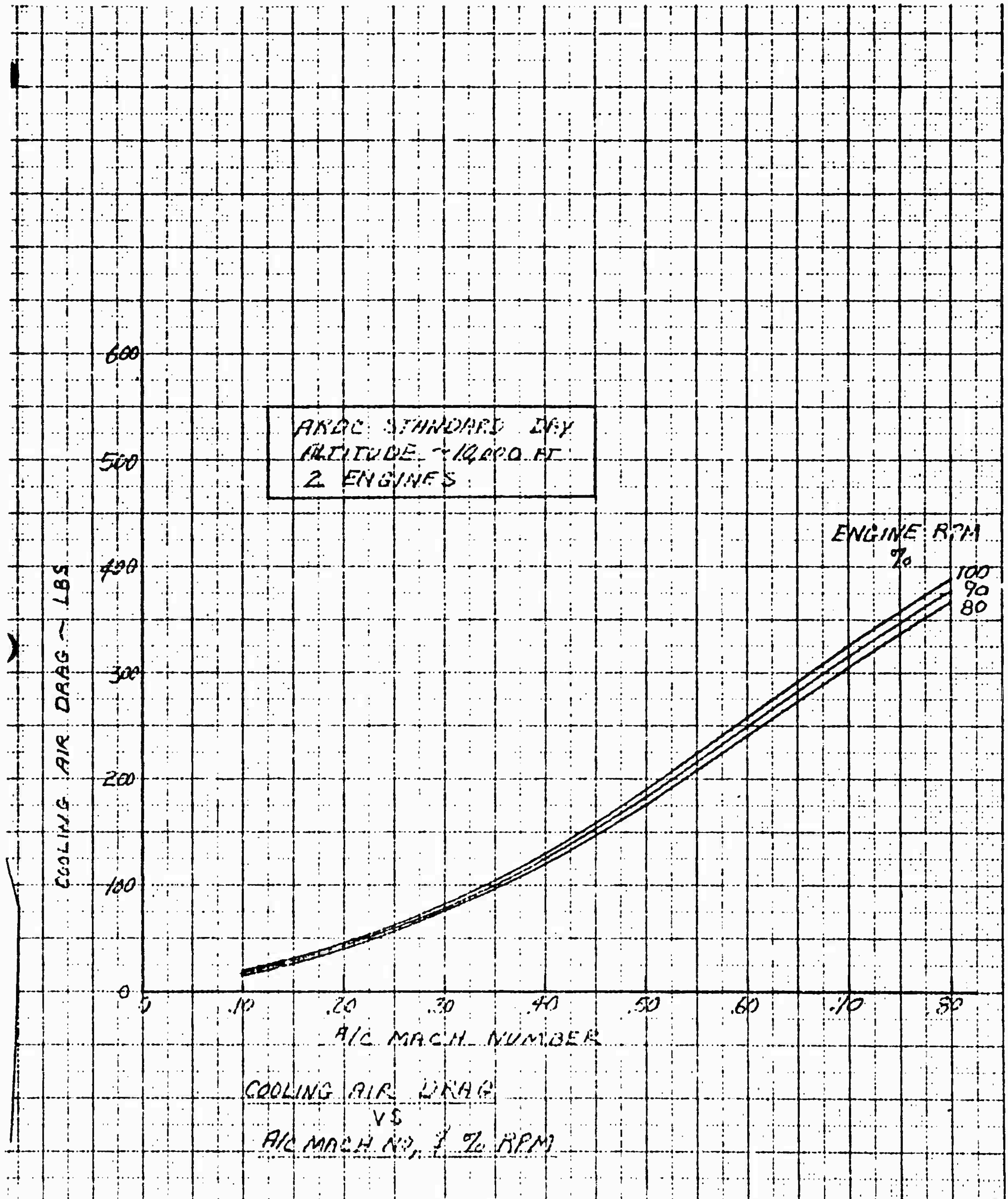

Figure 5.3 Cooling Air Drag vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Std. Day

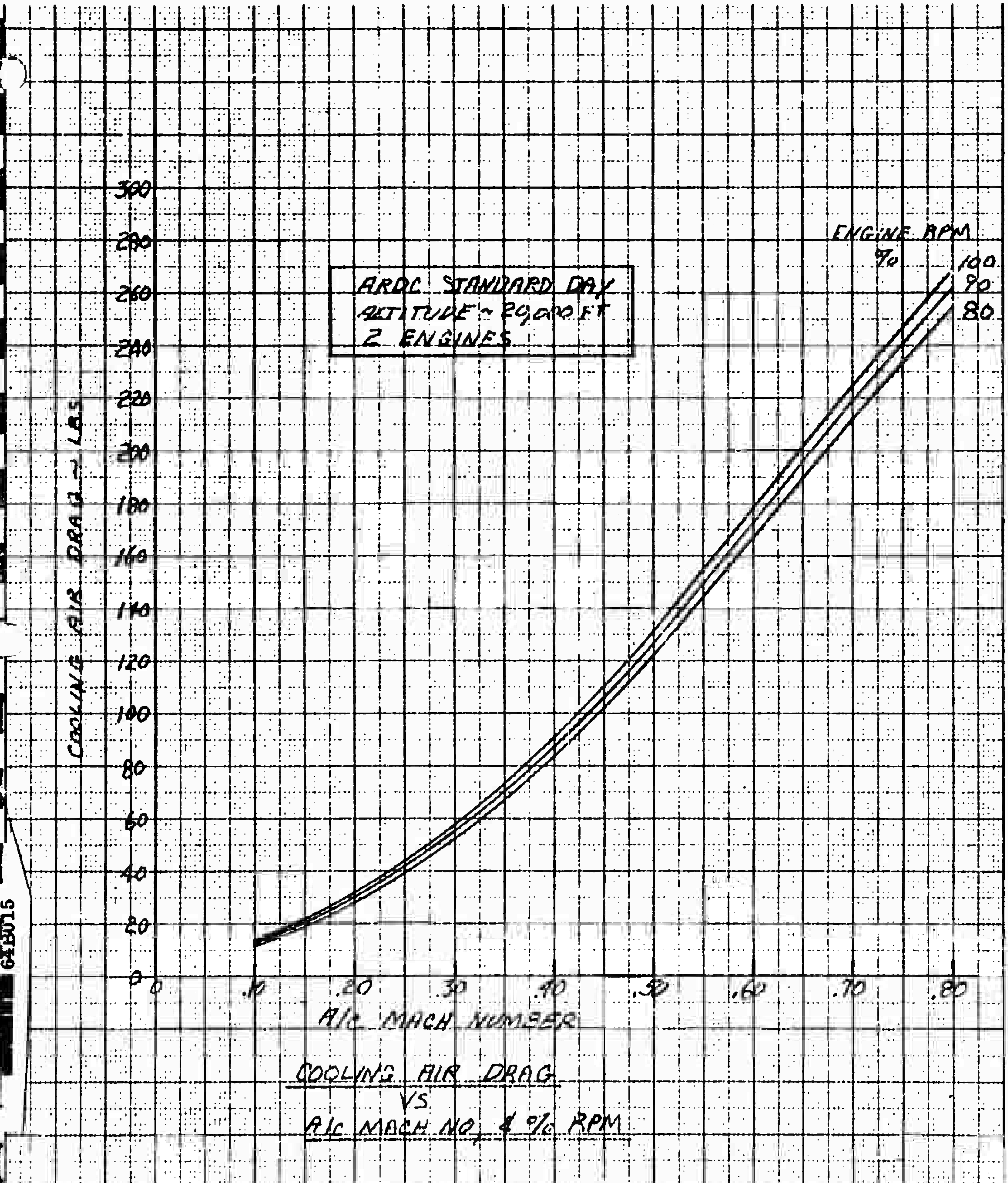

Figure 5.4 Cooling Air Drag vs Mach No. and % RPM, Altitude = 20,000 ft., 2 Engines, Std. Day

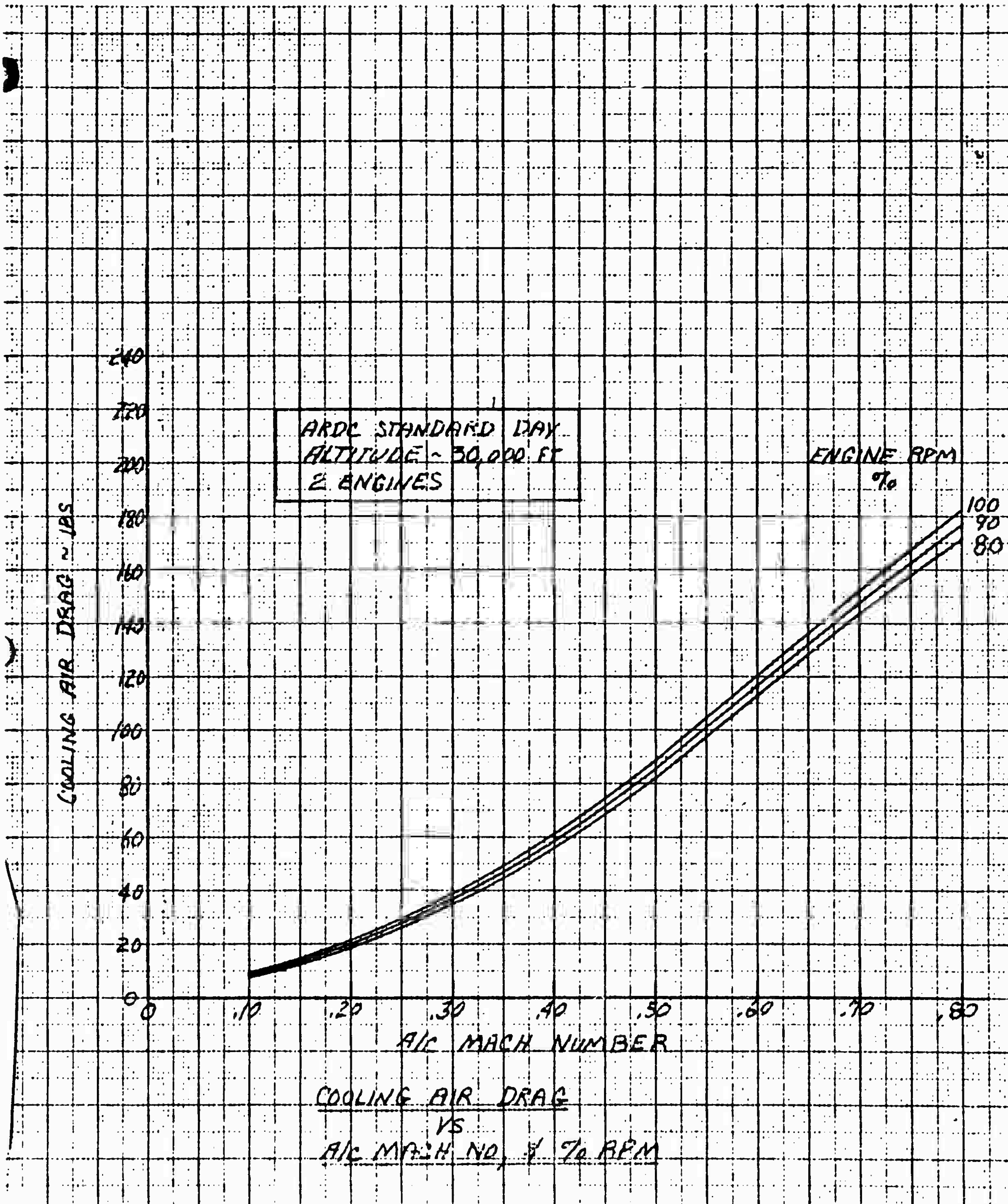

Figure 5.5 Cooling Air Drag vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Std. Day

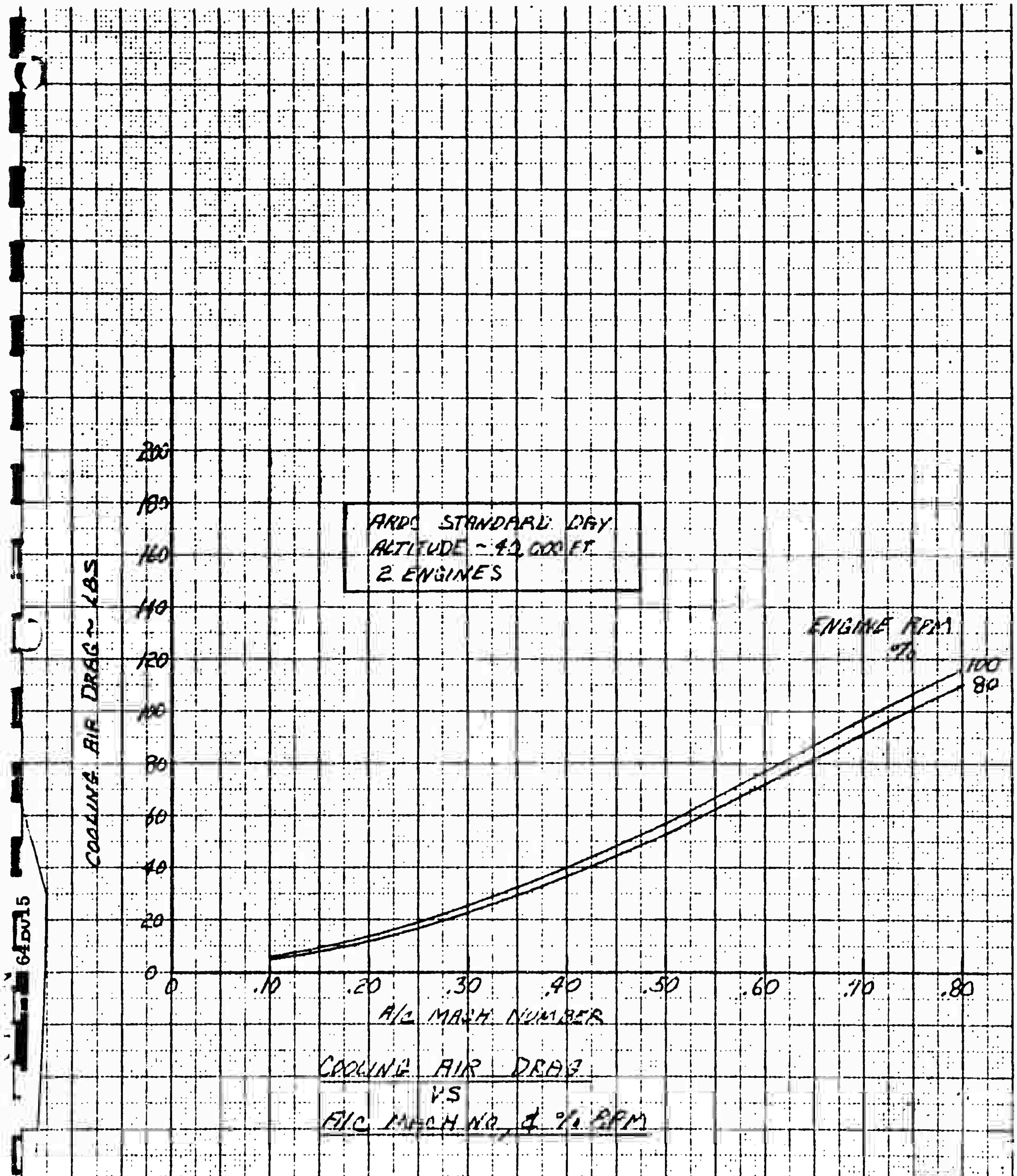

Figure 5.6 Cooling Air Drag vs Mach No. and % RPM, Altitude = 40,000 ft., 2 Engines, Std. Day

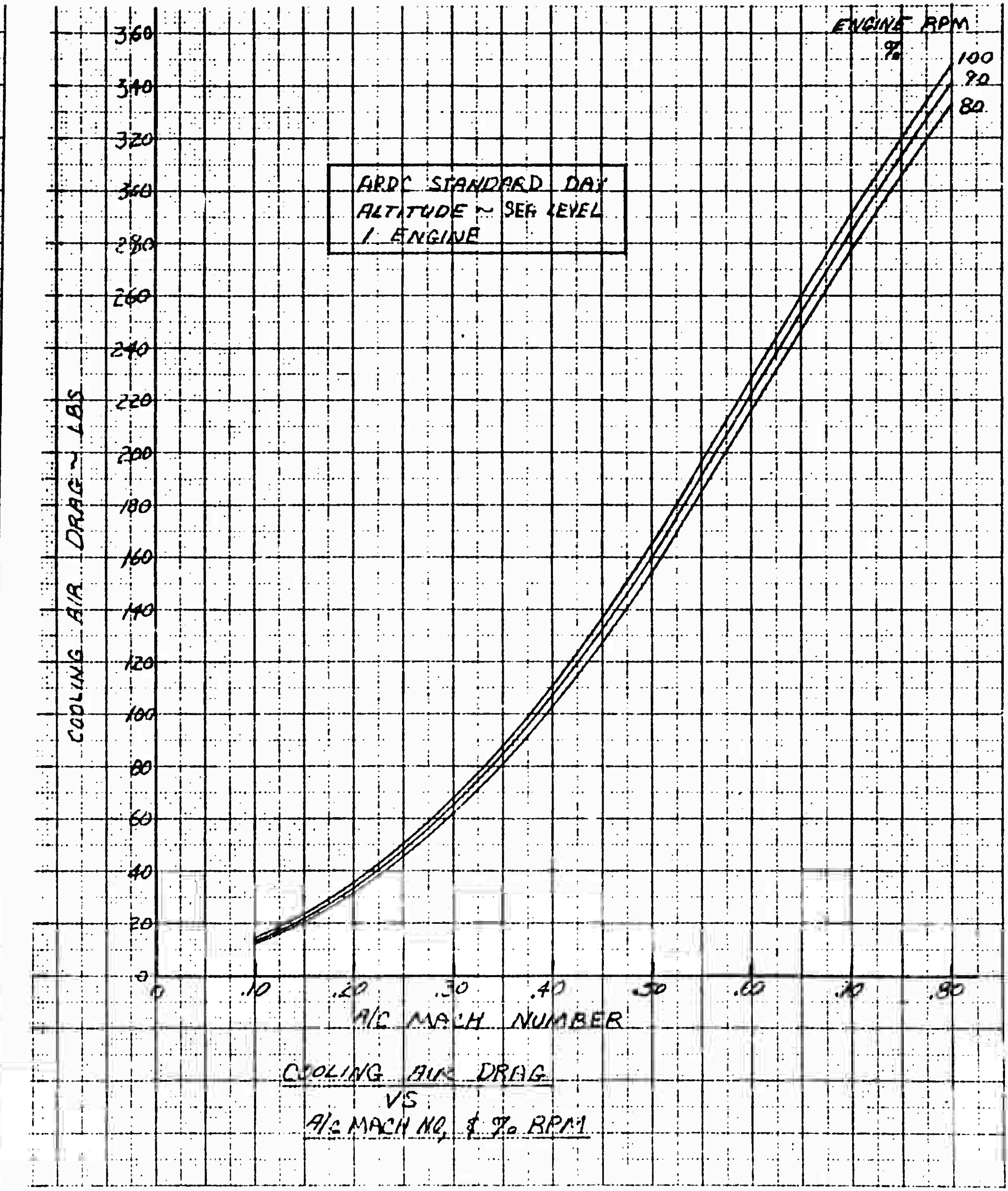

Figure 5.7 Cooling Air Drag vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Std. Day

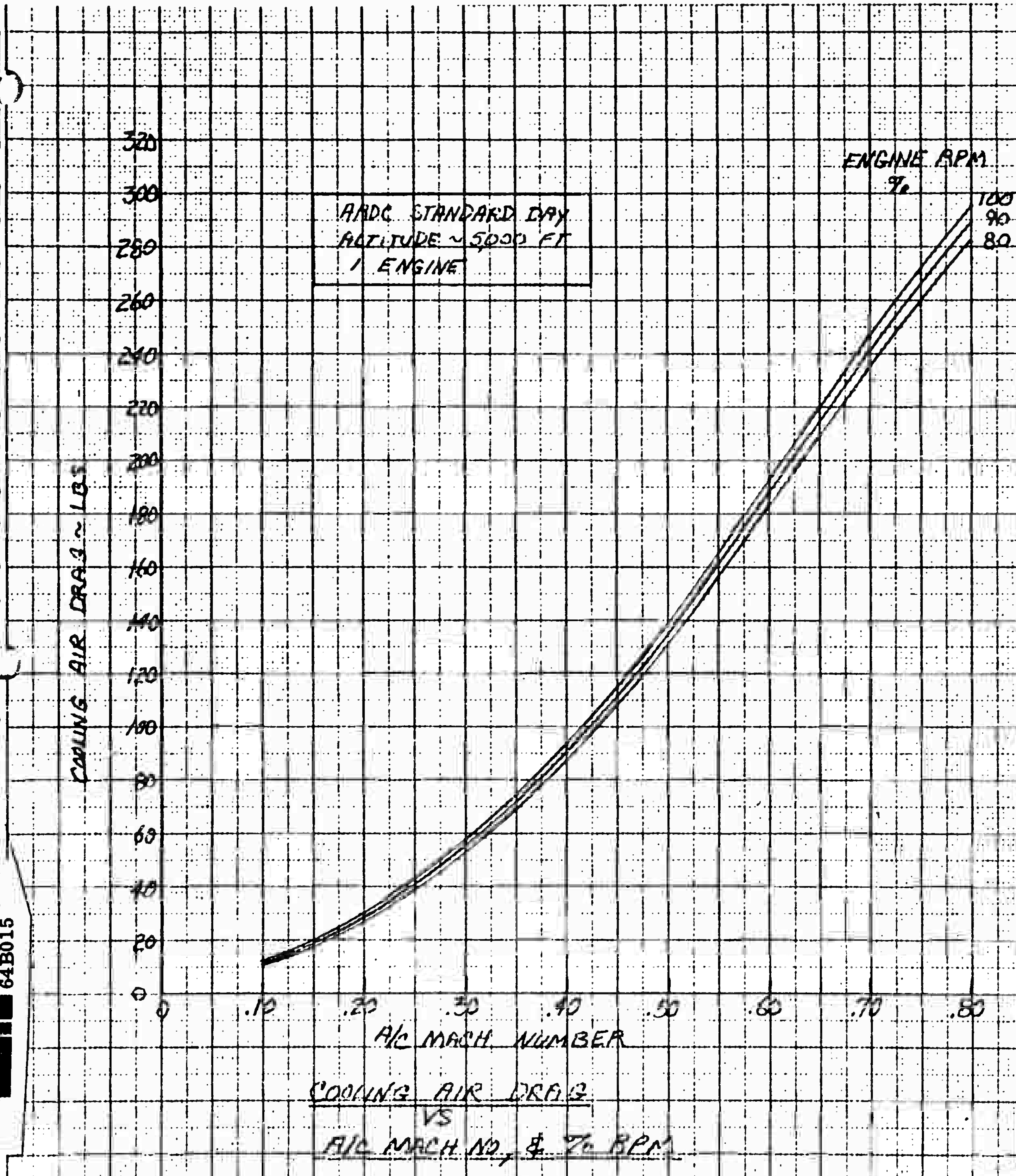

Figure 5.8 Cooling Air Drag vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Std. Day

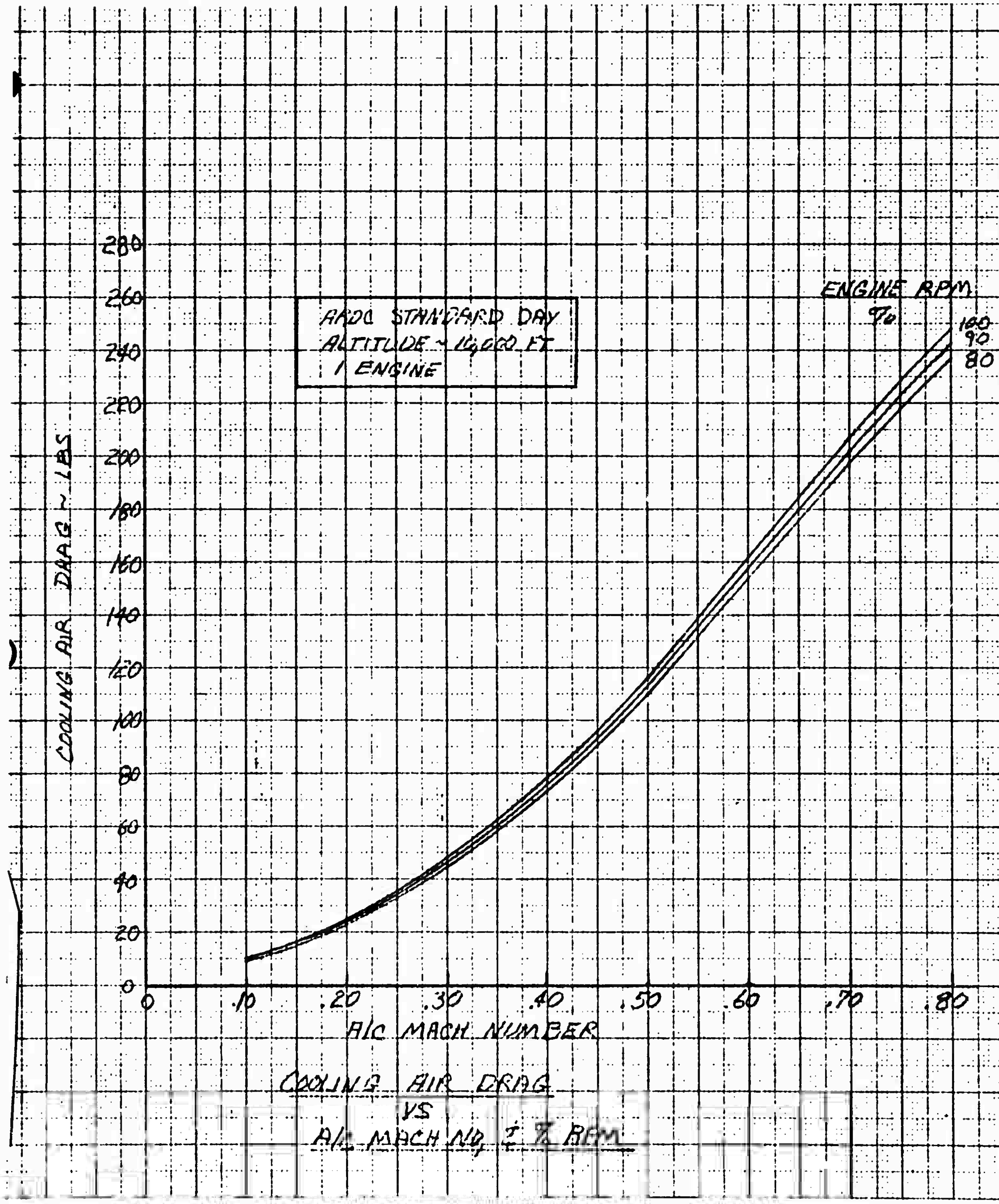

Figure 5.9 Cooling Air Drag vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Std. Day

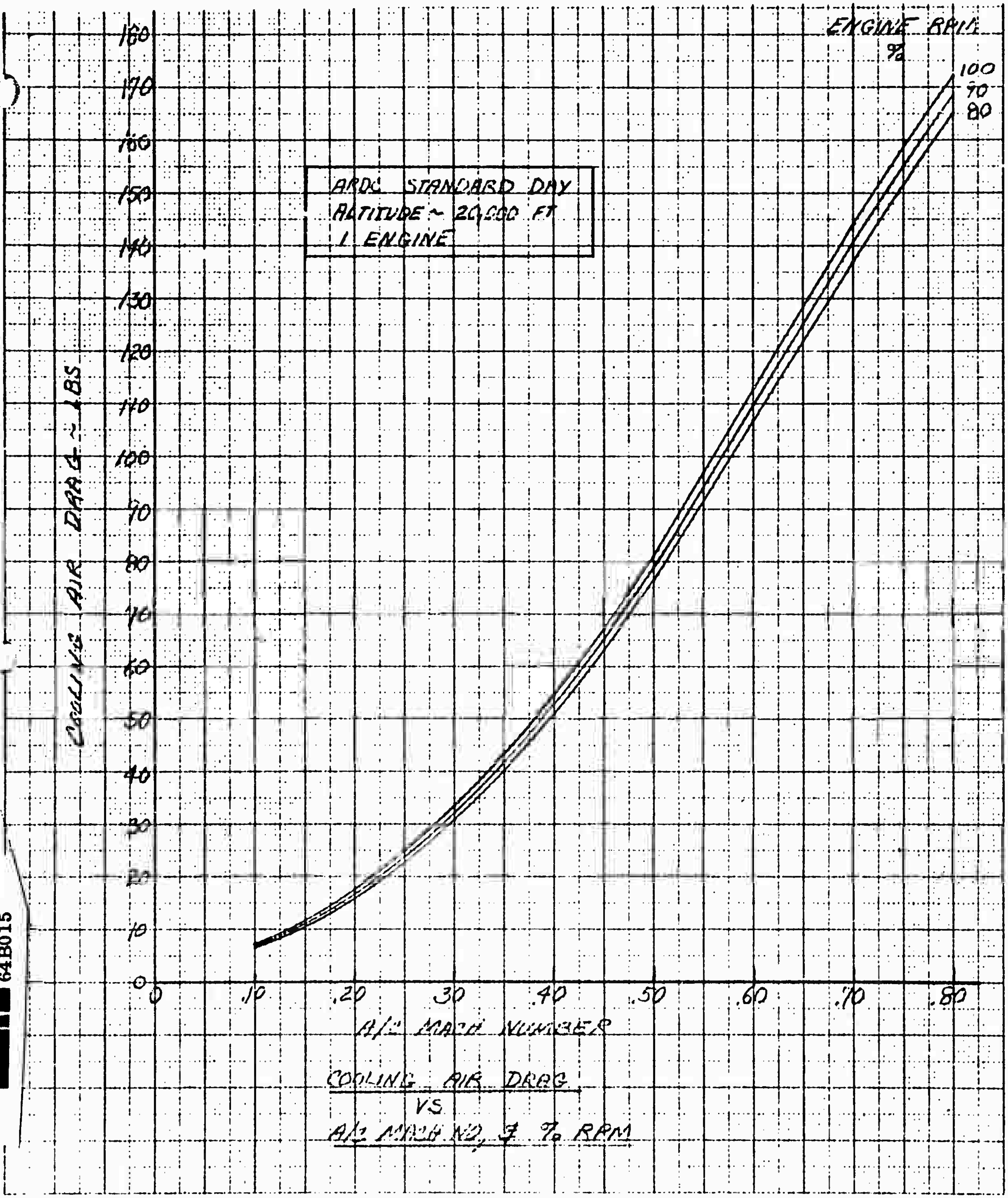

Figure 5.10 Cooling Air Drag vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Std. Day

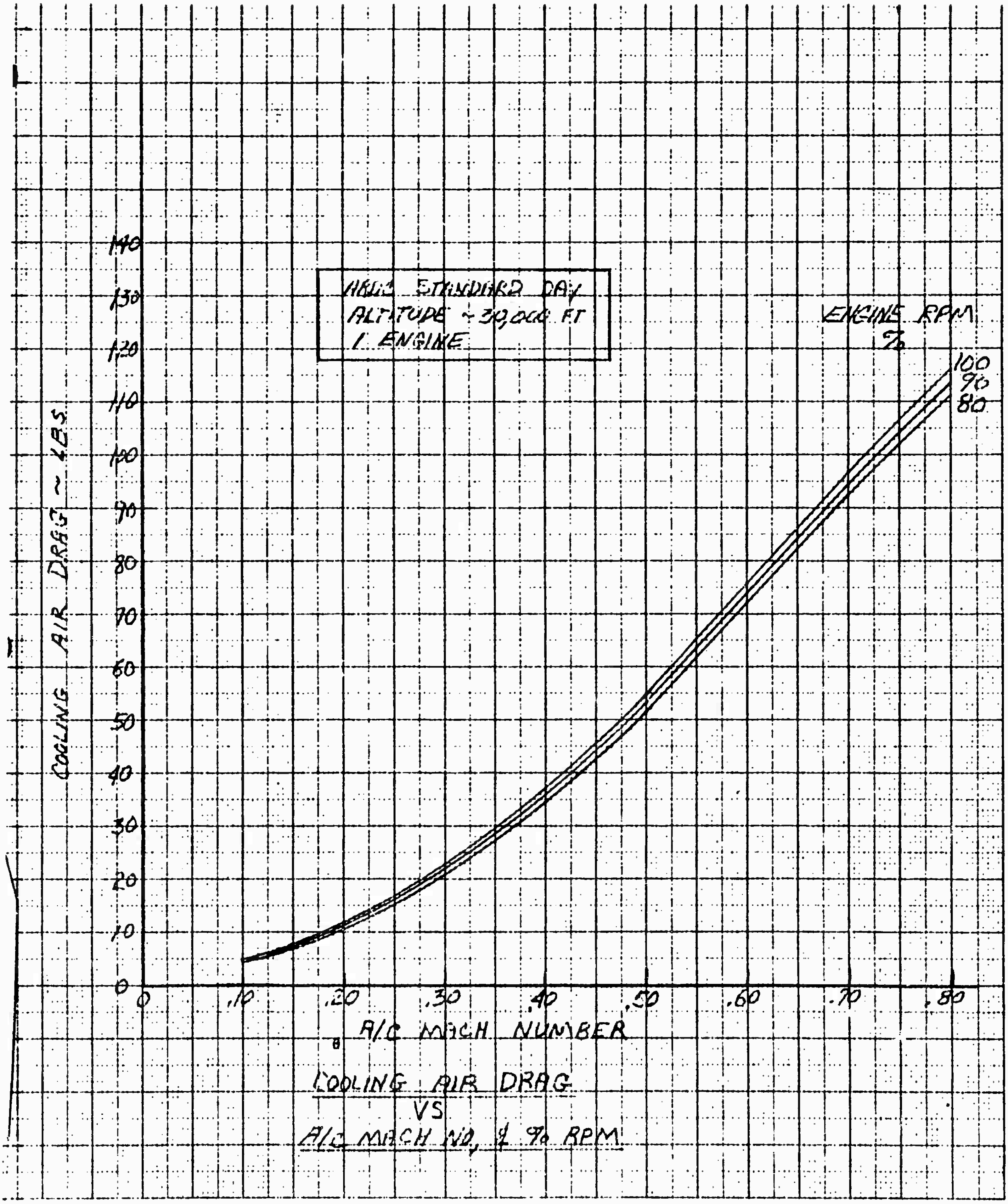

Figure 5.11 Cooling Air Drag vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Std. Day

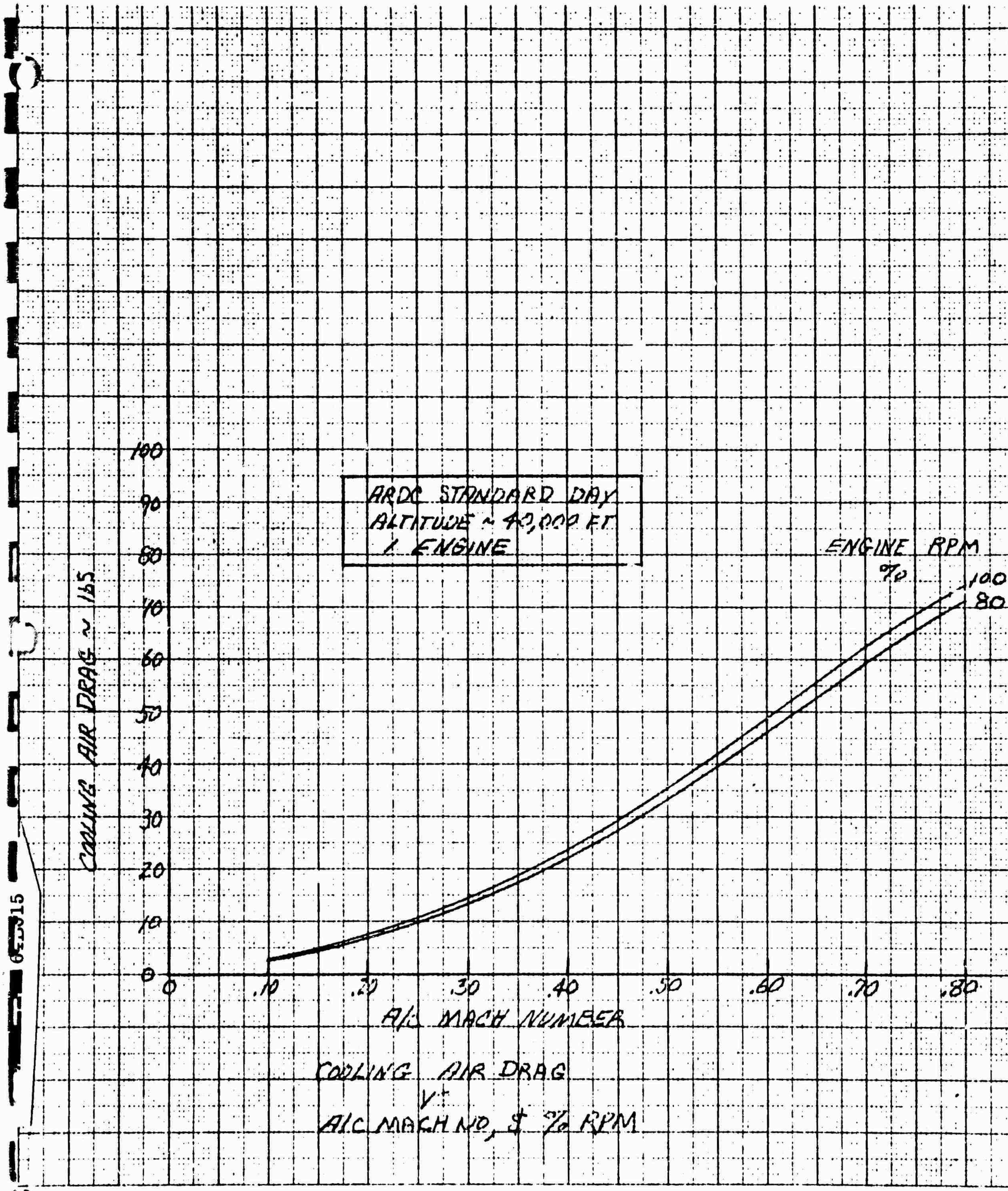

Figure 5.12 Cooling Air Drag vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Std. Day

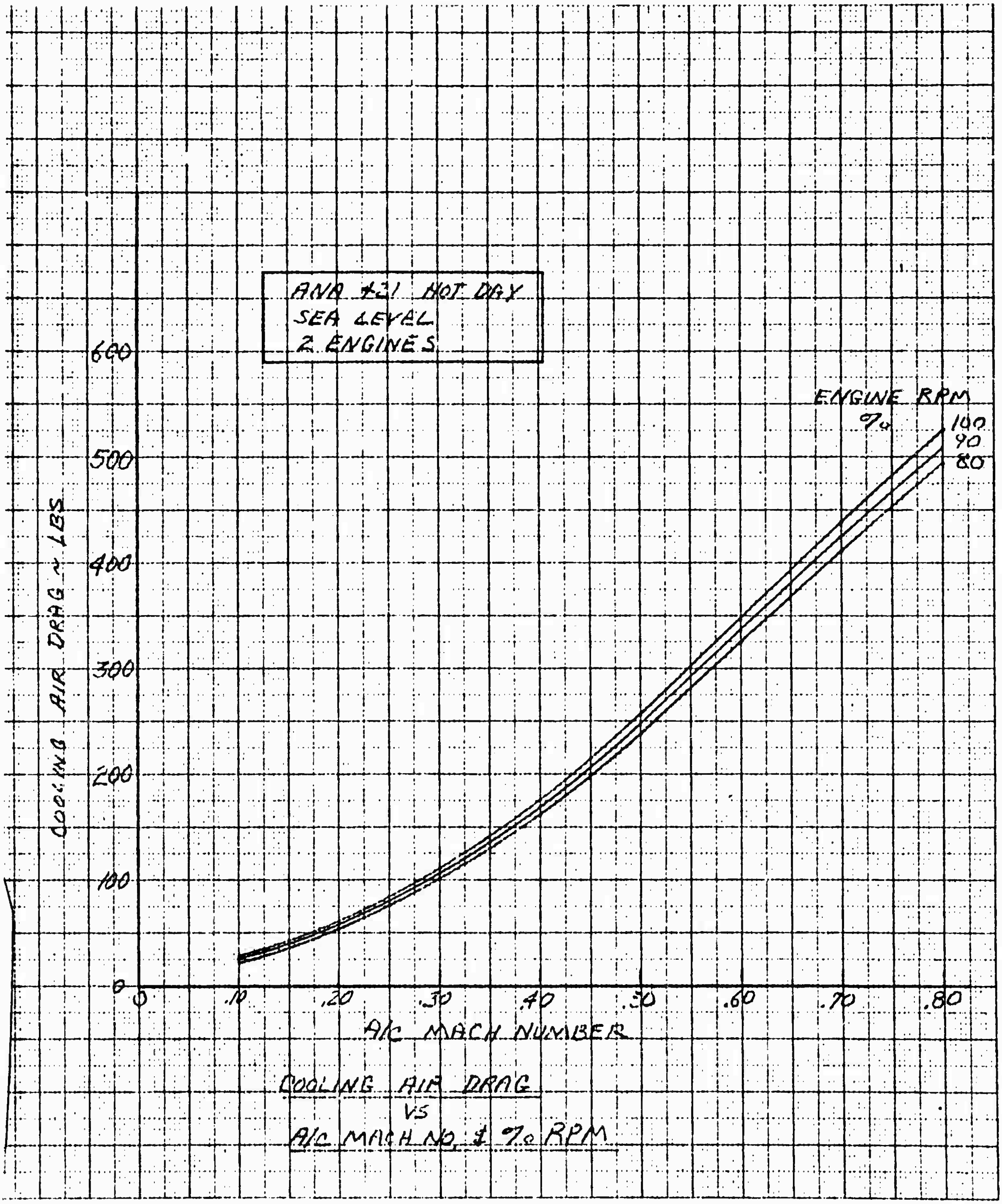

Figure 5.13 Cooling Air Drag vs Mach No. and % RPM; Altitude = 0 ft., 2 Engines, Hot Day

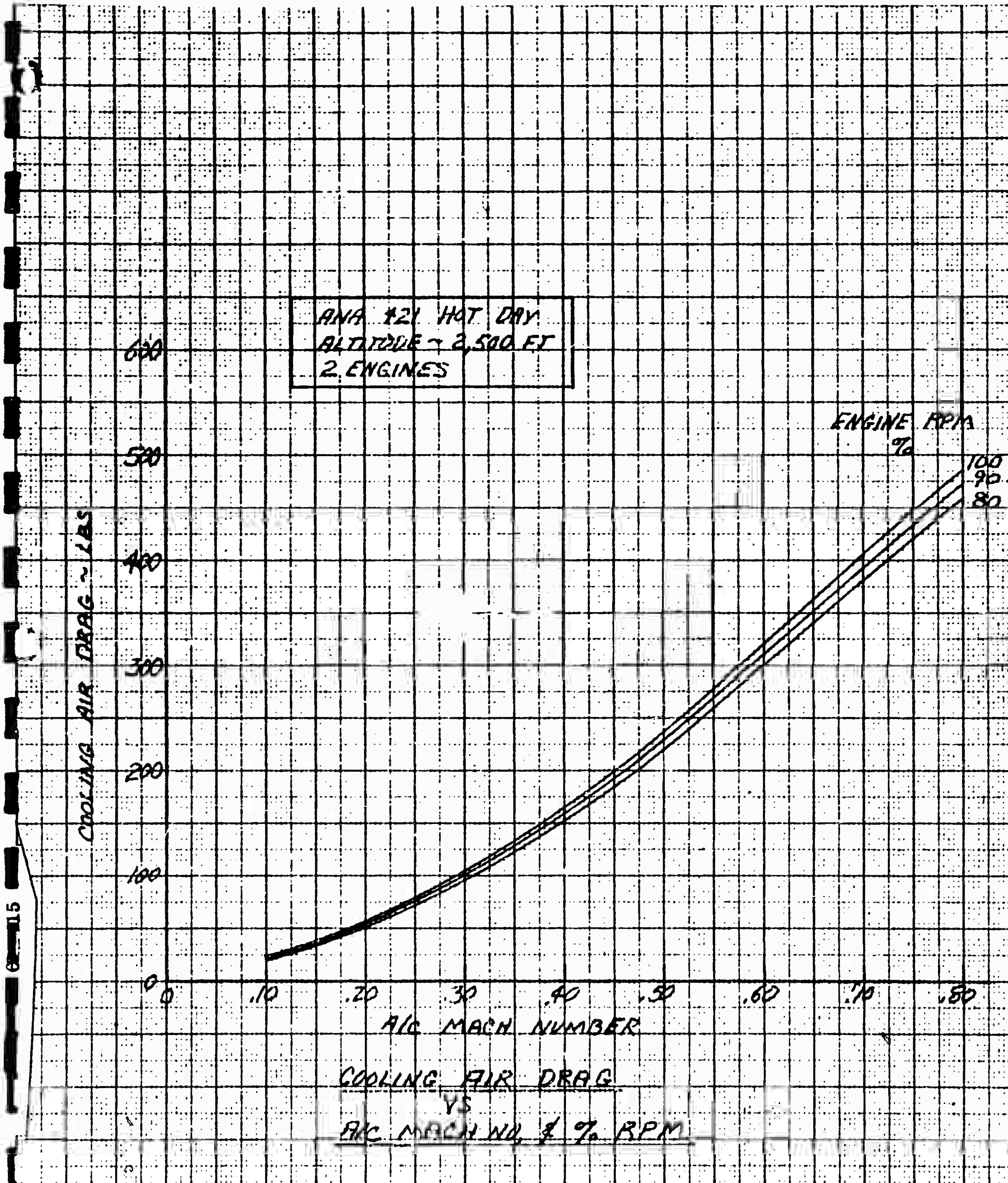

Figure 5.14 Cooling Air Drag vs Mach No. and % RPM; Altitude = 2500 ft., 2 Engines, Hot Day

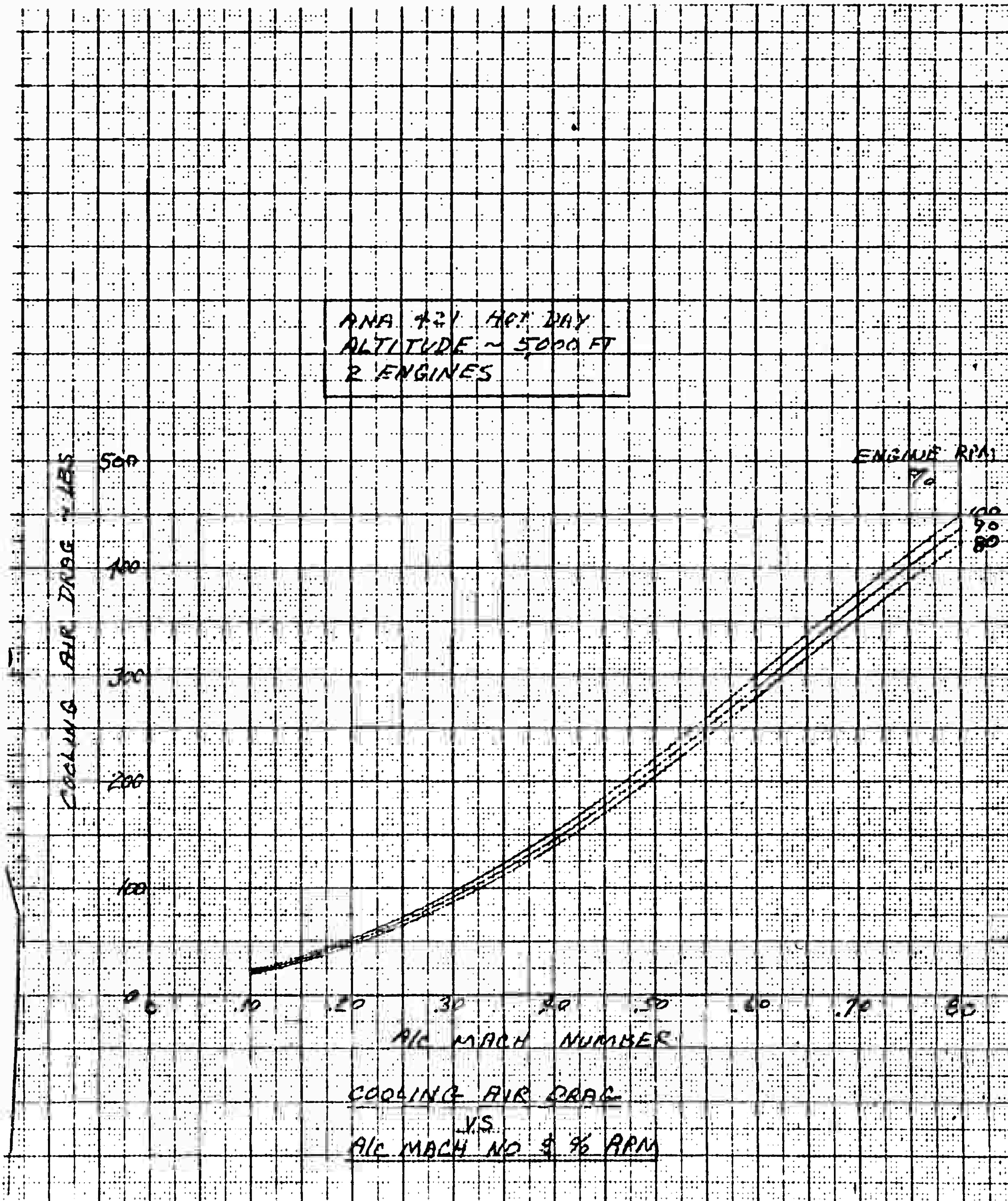

Figure 5.15 Cooling Air Drag vs Mach No. and % RPM; Altitude = 5000 ft., 2 Engines, Hot Day

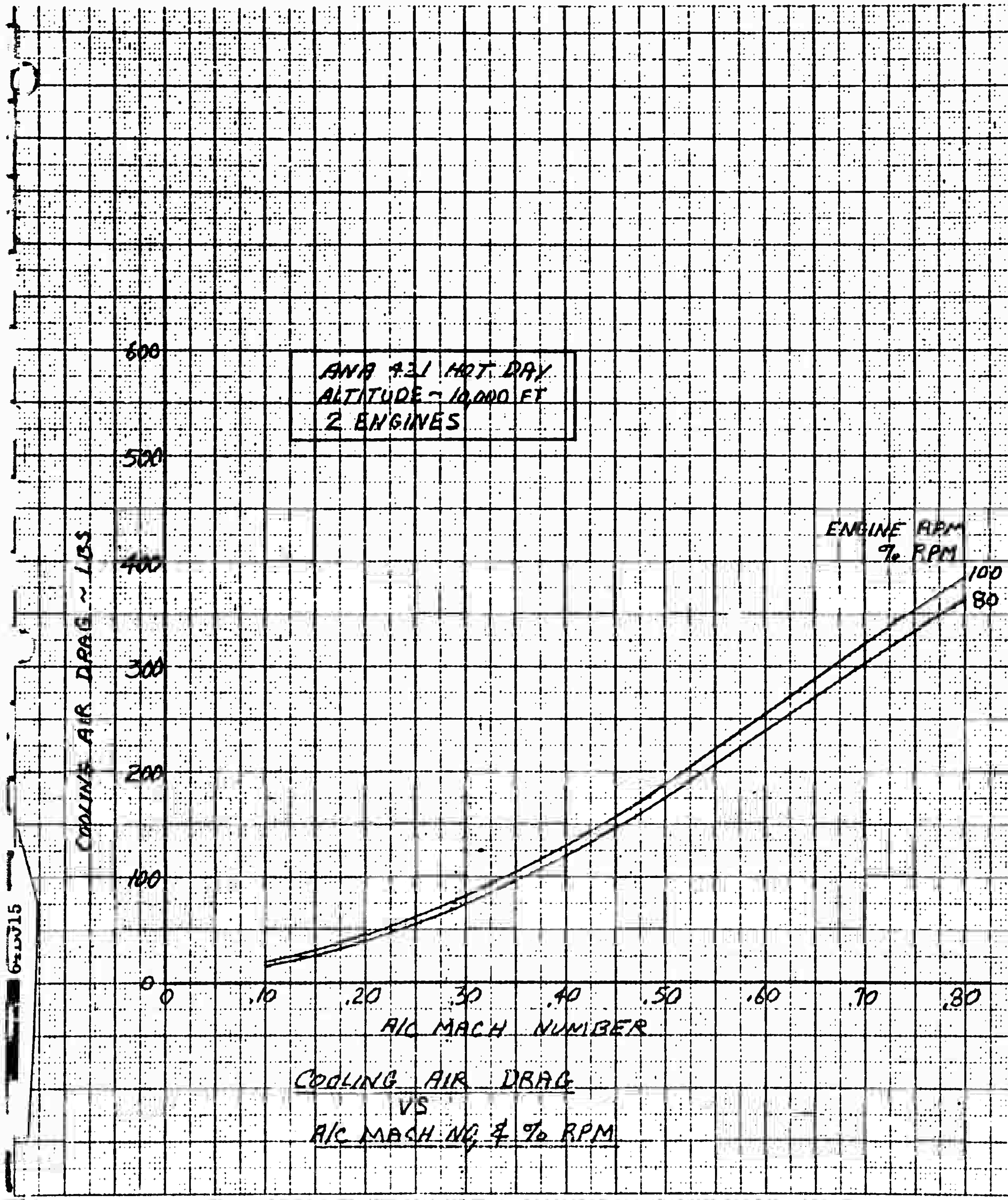

Figure 5.16 Cooling Air Drag vs Mach No. and % RPM; Altitude = 10,000 ft., 2 Engines, Hot Day

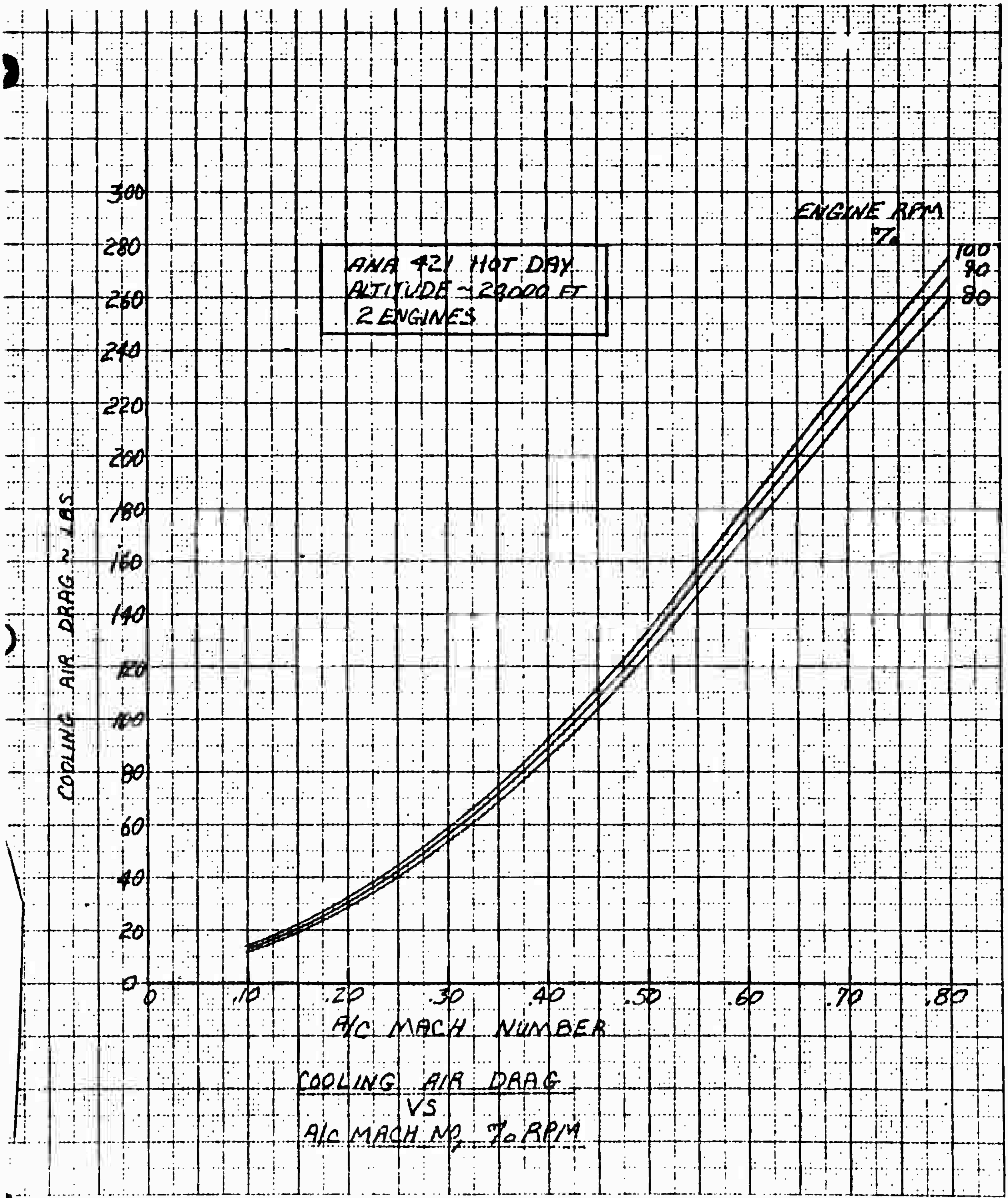

Figure 5.17 Cooling Air Drag vs Mach No. and % RPM; Altitude = 20,000 ft., 2 Engines, Hot Day

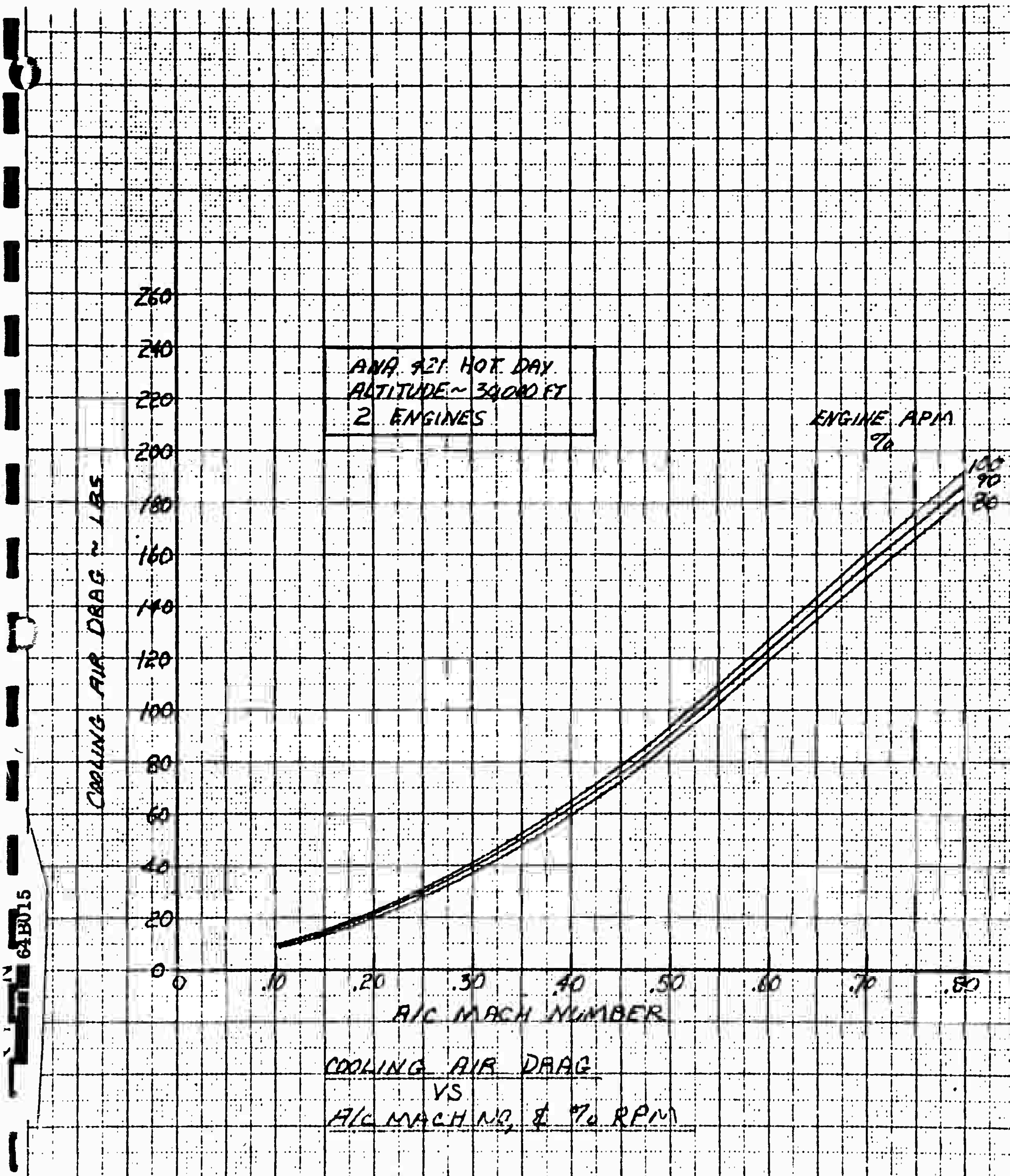

Figure 5.18 Cooling Air Drag vs Mach No. and % RPM; Altitude = 30,000 ft., 2 Engines, Hot Day

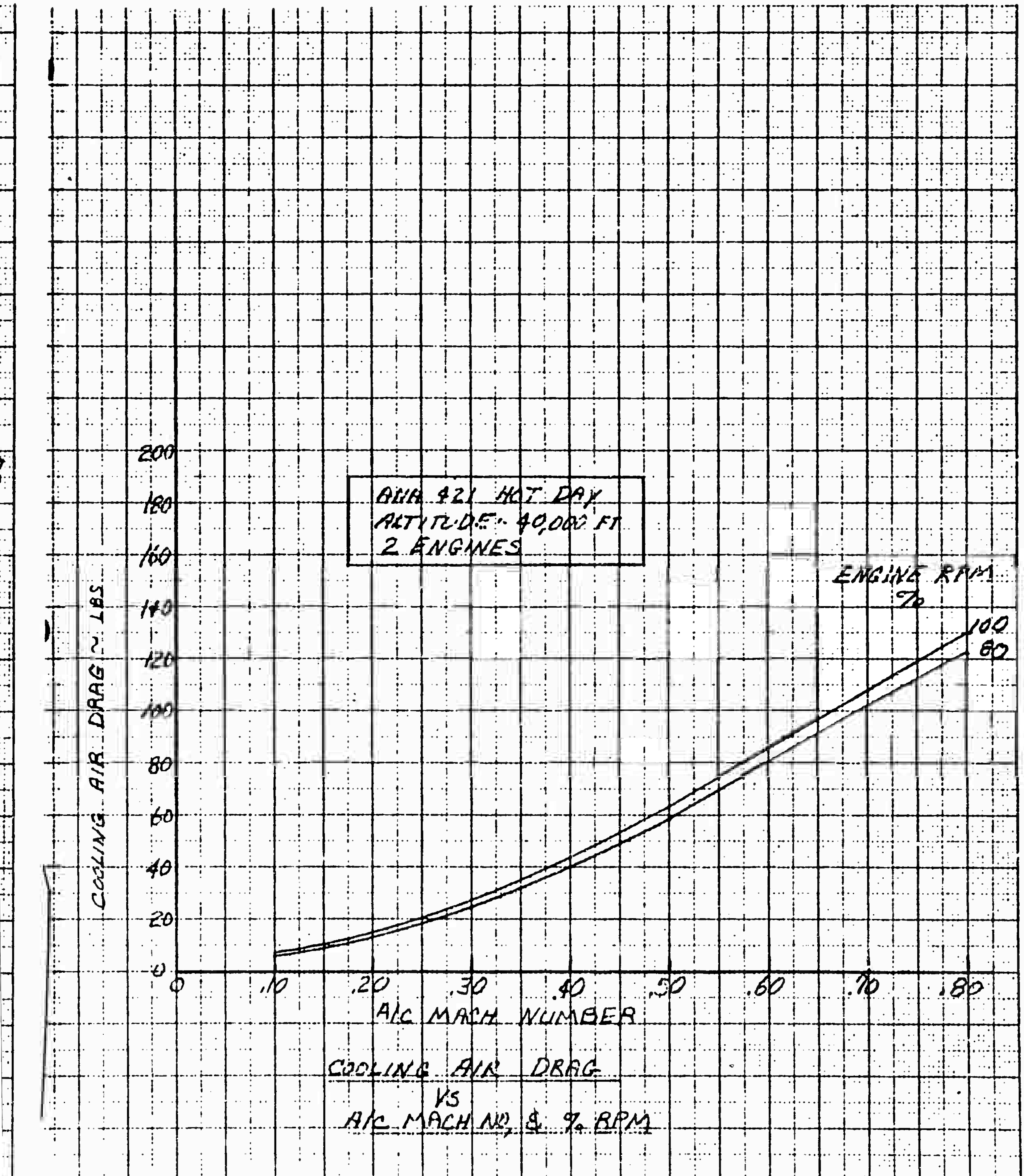

Figure 5.19 Cooling Air Drag vs Mach No. and % RPM; Altitude = 40,000 ft., 2 Engines, Hot Day

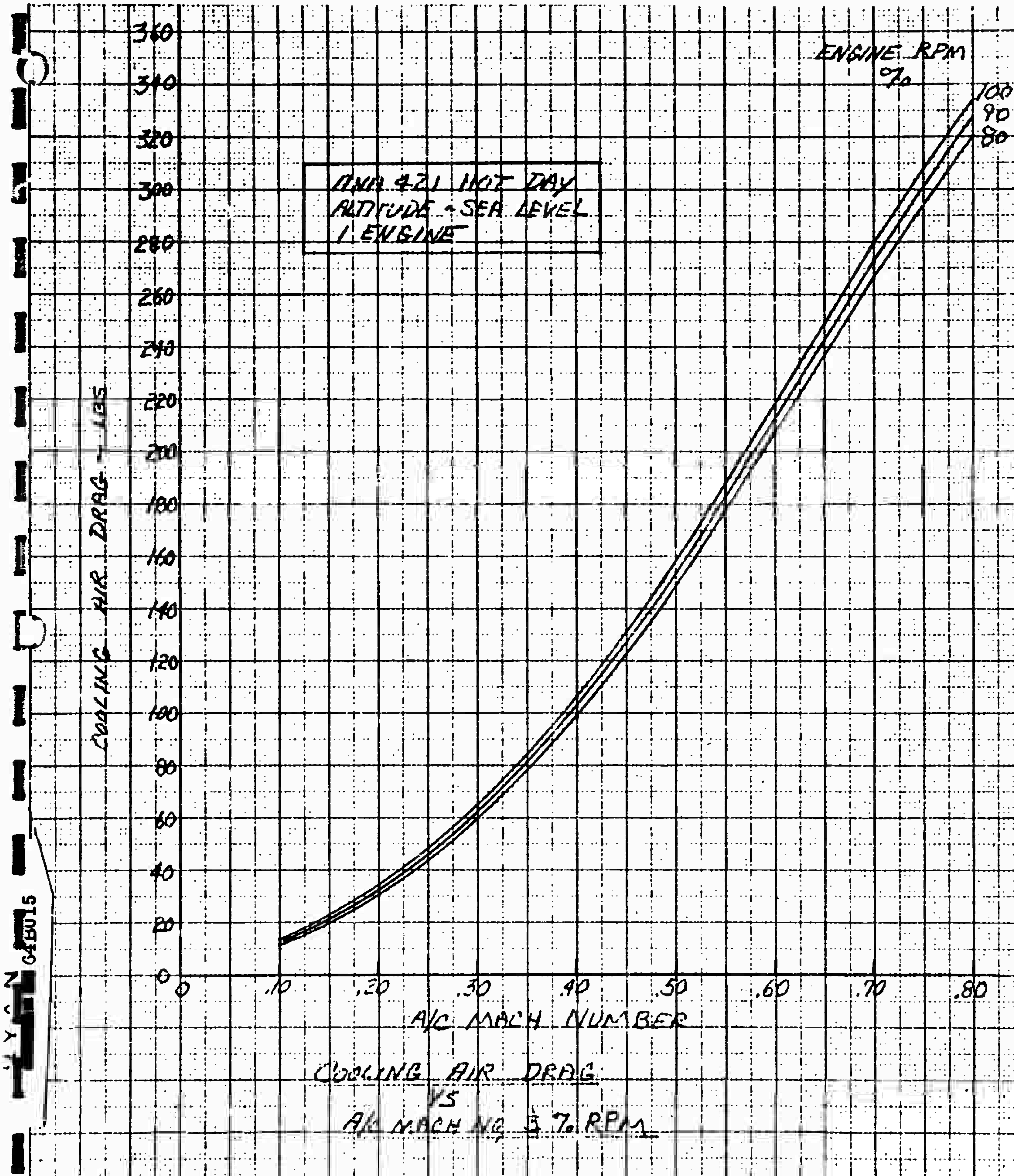

Figure 5.20 Cooling Air Drag vs Mach No. and % RPM; Altitude = 0 ft., 1 Engine, Hot Day

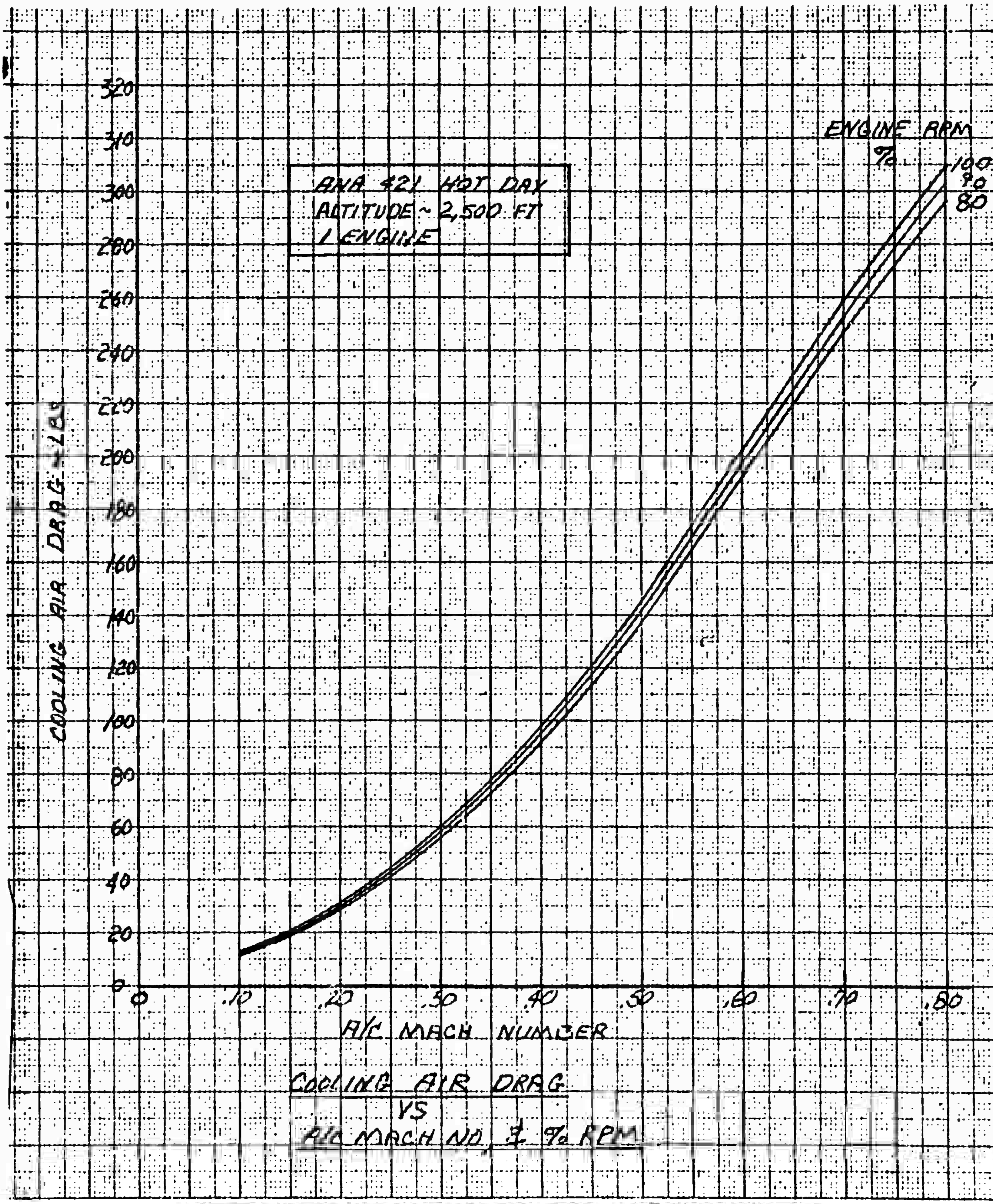

Figure 5.21 Cooling Air Drag vs Mach No. and % RPM; Altitude = 2500 ft., 1 Engine, Hot Day

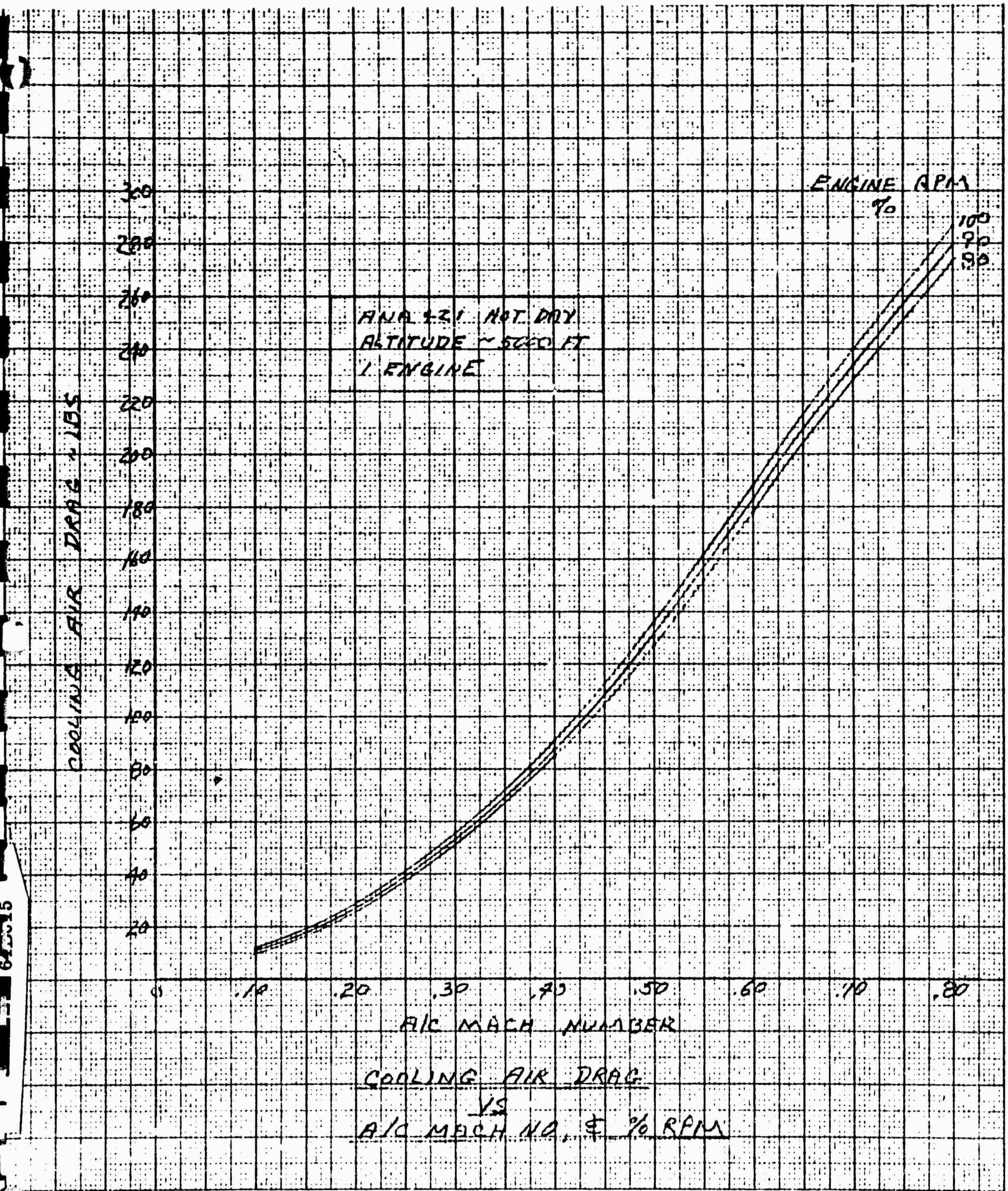

Figure 5.22 Cooling Air Drag vs Mach No. and % RPM; Altitude = 5000 ft., 1 Engine, Hot Day

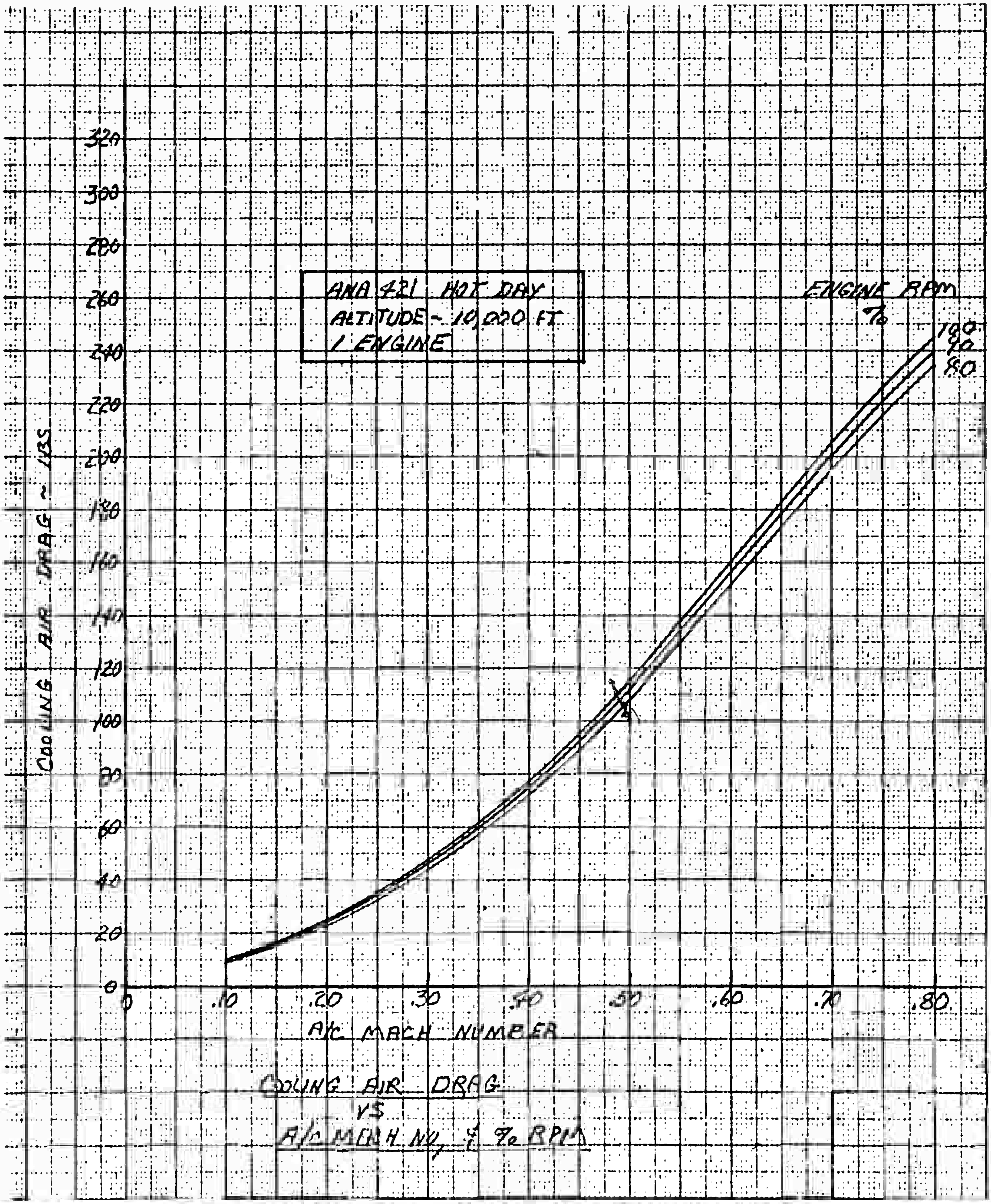

Figure 5.23 Cooling Air Drag vs Mach No. and % RPM; Altitude = 10,000 ft., 1 Engine, Hot Day

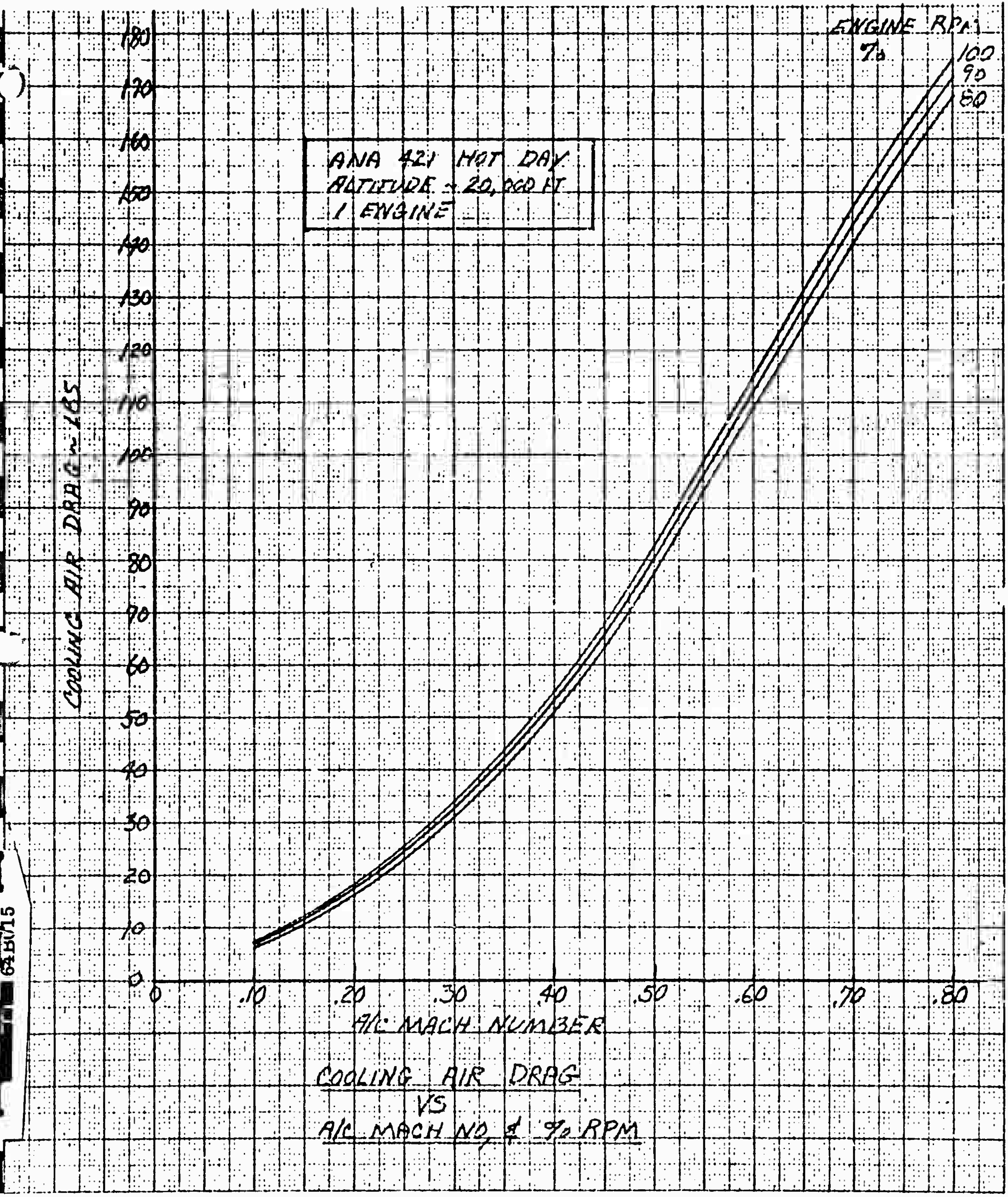

Figure 5.24 Cooling Air Drag vs Mach No. and % RPM; Altitude = 20,000 ft., 1 Engine, Hot Day

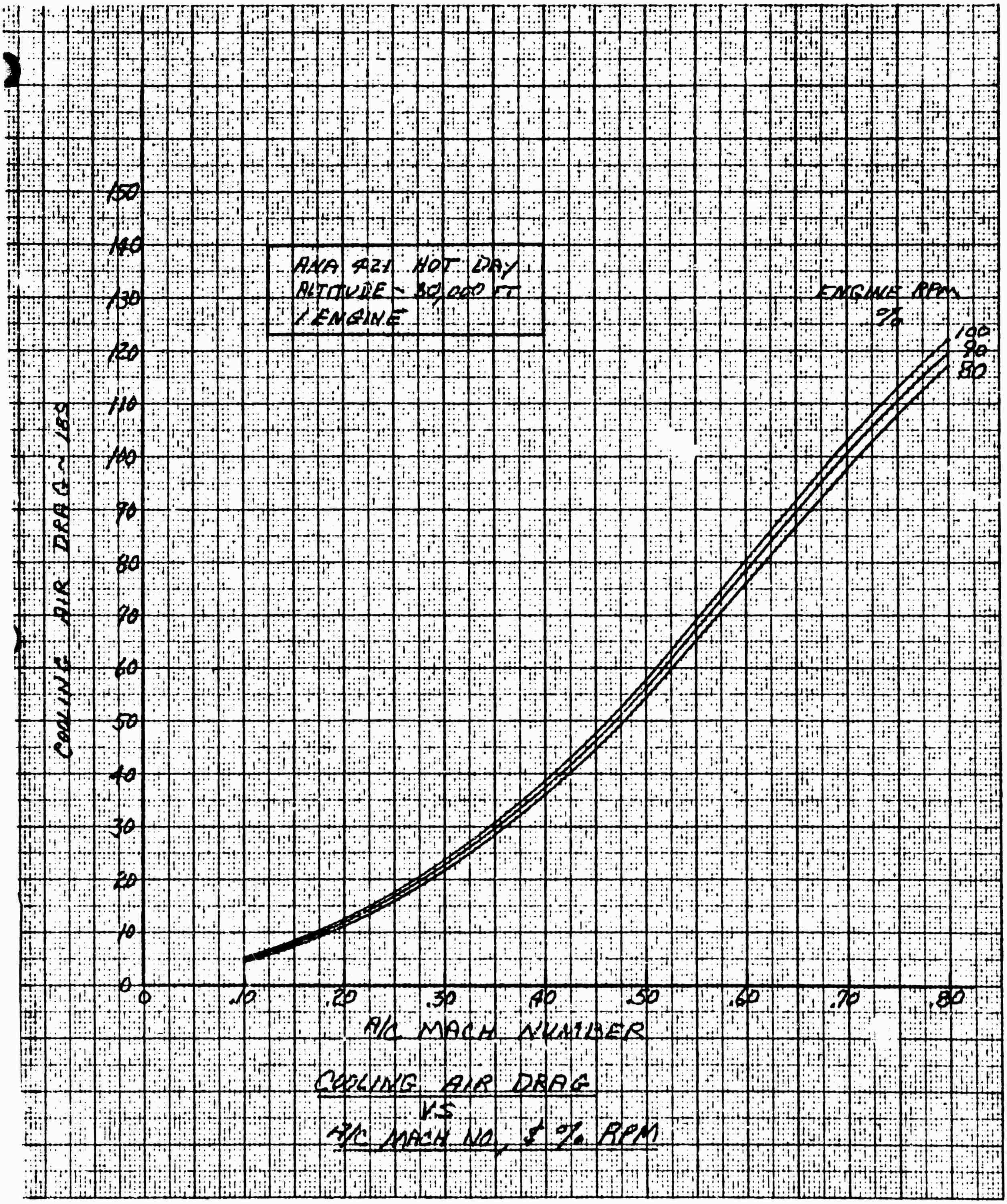

Figure 5.25 Cooling Air Drag vs Mach No. and % RPM; Altitude = 30,000 ft., 1 Engine, Hot Day

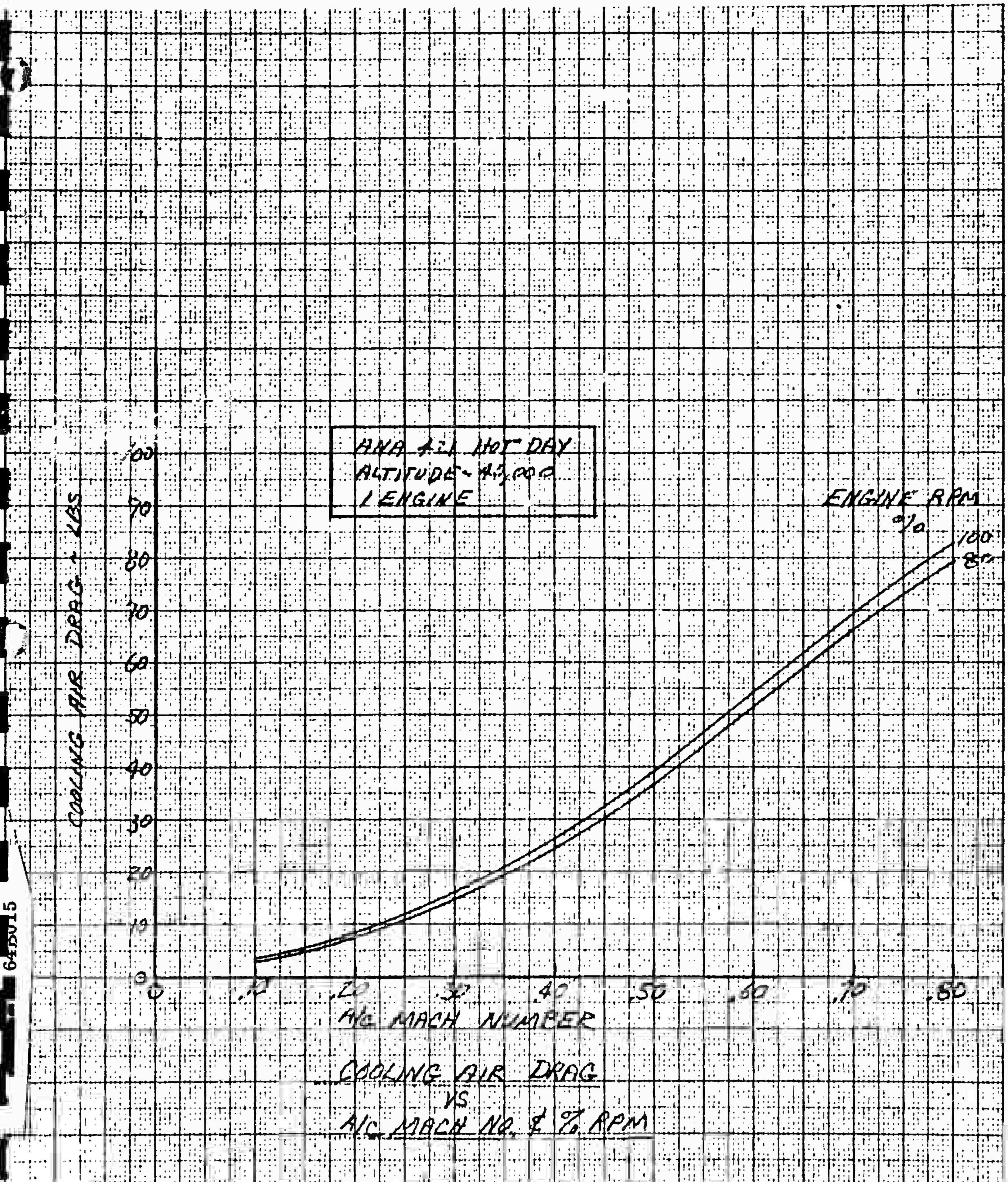

Figure 5.26 Cooling Air Drag vs Mach No. and % RPM; Altitude = 40,000 ft., 1 Engine, Hot Day

6.0 CONCLUSIONS AND RECOMMENDATIONS

1. Calculated installed fan mode performance based on model specification data could not be used to predict aircraft fan mode operation except for zero speed hovering conditions out of ground effect. Total untrimmed static lift with no allowance for control requirements is 15,020 pounds for an ARDC Standard Day at sea level; and 12,625 pounds for an ANA 421 Hot Day at 2500 feet.

2. Current state-of-the-art does not permit separation of lift-fan propulsion system and airframe contributions to aircraft performance. This requires that XV-5A transition performance be derived, using lift-fan power coefficients obtained from wind tunnel model tests and correlated in terms of propulsion system and aircraft operating conditions.

3. A systematic study of lift fan propulsion system and aircraft variables is recommended in order to isolate and establish the respective contributions to, and effects on, aircraft performance; and to develop methods for their prediction and correlation. Its scope should be sufficient to establish scale, transition speed, geometry, and ground effects on aircraft performance.

4. A systematic study of factors affecting locally induced environments of the aircraft is recommended in order to develop a detailed knowledge and understanding of the flow fields around the aircraft as they affect hot gas re-ingestion, propulsion system performance, external surface heating, downwash particle impingement and particle re-ingestion phenomena, and stability and control characteristics.

5. Calculated turbojet mode installed performance as presented in this report must be limited to the flight-altitude envelope of the model specification. Estimated sea level static thrust of the aircraft is 4920 pounds for an ARDC Standard Day. For an ANA 421 Hot Day at 2500 feet, the static thrust is 4250 pounds.

6. The presence of substantial hot gas re-ingestion by the J85 gas generator at near idle power (47 to 60% RPM) when in proximity to the ground, combined with power extraction from the gas generator causes the exhaust gas temperatures to exceed their allowable limits, therefore, a minimum J85 gas generator rpm of 70% is recommended for XV-5A fan mode operation.

7. Performances of the ejector and thrust spoiler are adequate to perform their intended functions.

8. The J85 gas generator air inlet shows excellent performance over its required operating range. Inlet model test data suggest total pressures at large sideslip angles and low speeds may exceed slightly the gas generator limits of 10%, however in the actual inlet, internal lines were modified slightly in the region of maximum distortion to reduce the diffusion angle; thus, actual variations should be within allowable limits. The inlet provides a minimum static pressure recovery of 98.9 percent and the pressure recovery exceeds 99 percent for high speed cruise flight (Mach 0.70).

9. The propulsion system and its associated components should perform intended functions adequately throughout the XV-5A flight envelope.

7.0 APPENDIX

7.1 REFERENCES

1 General Electric Company, Flight Propulsion Laboratory Dept., Cincinnati, Ohio, X353-5B Propulsion System Specification, Specification No. 112 - January 15, 1962; Revised October 23, 1962.

2 General Electric Company, Flight Propulsion Laboratory Dept., Cincinnati, Ohio, Instruction Manual Customer Computer Deck, GE X353-5B Lift Fan System Performance (Specification No. 112), January 1962.

3 General Electric Company, Flight Propulsion Laboratory Dept., Cincinnati, Ohio, Customer Computer Deck for GE X353-5B Lift Fan System Performance (Specification No. 112), January 1962.

4 General Electric Company, Flight Propulsion Laboratory Dept., Cincinnati, Ohio, X376 Pitch Fan Specification, Specification No. 113, March 1, 1962; Revised October 23, 1962.

5 General Electric Company, Flight Propulsion Laboratory Dept., Cincinnati, Ohio, Instruction Manual, Customer Computer Deck for GE-X376 Pitch Fan System Performance, April 1962.

6 General Electric Company, Flight Propulsion Laboratory Dept., Cincinnati, Ohio, Customer Computer Deck for GE-X376 Pitch Fan System Performance, April 1962.

7 Estimated Static Stability and Control Characteristics, General Electric Company, Report No. 146.

8 Letter E. G. Smith (G.E.) to H. B. Starkey/W. B. Davis (Ryan), System Static Performance X353-5B and X376, (Reproduced in Appendix 7.2), May 15, 1963.

9 General Electric Company, Flight Propulsion Laboratory Dept., Cincinnati, Ohio, Instruction Manual Customer Computer Deck for General Electric X353-5B Turbojet Mode Performance (Specification No. 112), January 1962.

10 General Electric Company, Flight Propulsion Dept., Cincinnati, Ohio, Customer Computer Deck for GE X353-5B Turbojet Mode Performance (Specification No. 112), January 1962.

11 B. W. Ela, Wind Tunnel Test Report of 1/5 Scale Inlet Model U.S. Army XV-5A Lift Fan Research Aircraft, in preparation.

12 R. E. Pendley, J. R. Milillo, F. F. Fleming, An Investigation of Three NACA 1-Series Nose Inlets at Subsonic and Transonic Speeds, NACA Report RML52J23, 7 January 1953.

13 W. K. Greathouse and D. P. Hollister, Preliminary Air Flow and Thrust Calibrations of Several Cooling-Air Ejectors with a Primary to Secondary Temperature Ratio of 1.0, NACA Report RME52E21, dated July 22, 1952.

14 B. W. Ela, C. G. Brenson, Calculated Heat Transfer and Cooling System Performance U.S. Army XV-5A Lift Fan Research Aircraft, in preparation.

15 General Electric Company, Systems Description and Operating Instructions J85-5B Non-After-burning Engine, (Undated and Unidentified).

16 G. Ernst, The Use of Turbojet Thrust Deflection for Airplane Flight Control, Paper Delivered at Association Technique Maritime et Aeronautique; I, boul. Haussmann, Paris, 1957 Session.

17 U. H. Von Glahn and J. H. Povolny, Considerations of Some Jet-Deflection Principles for Directional Control and for Lift, SAE Paper 219 Delivered at SAE National Aeronautic Meeting, Los Angeles, California, October 1957.

7.2 SYSTEM STATIC PERFORMANCE X-353-5B AND X-376

The following document is that noted in Reference 8, Section 7.0, and is reproduced in whole for the convenience of the reader.

A. W. Kuhns

E/C

SYSTEM STATIC PERFORMANCE

X-353-5B and X-376

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E. G. Smith

May 15, 1963

H. B. Starkey/W. B. Davis
XV-5A Program
Ryan Aerospace Division
Ryan Aeronautical Company
San Diego 12, California

System static performance of the X-353-5B and X-376 fan system has been evaluated for the XV-5A installation. The performance is summarized in the attached figures for a range of altitudes at both standard day and hot day conditions. The sources of data and assumed installation effects included in the performance are as follows.

1. Average main and pitch fan performance based on Flightworthiness and Acceptance Tests of first three main and pitch fans.

2. Minimum J-85 gas generator performance as presently included in X-353-5B and X-376 specifications and customer deck.

3. Main fan exit louver vector and stagger angles, β_v and β_s , are zero degrees.

4. Nose fan thrust modulator is set at the maximum vertical lift position.

5. Gas generator performance corrected for inlet performance and power extraction using the following Ryan furnished data:
   - inlet recovery, $\eta_r = 0.984$
   - horsepower extraction, HPE = 58 per engine
   - compressor bleed, WB3 = .022 #/sec. per engine.

6. Main fan and pitch fan inlet losses as obtained during scale model tests using the 26" LSSM fan.

7. No main fan, pitch fan or gas generator hot gas reingestion.

8. No ground effects (height greater than 3 fan diameters).

9. Diverter valve, cross duct and pitch fan duct losses as measured during the Flightworthiness and Acceptance Tests.

10. Pitch fan scroll areas set for 12.3 percent bleed. This bleed setting is maximum permissible without causing pitch fan overspeed problems and at the same time produce an installed lift in excess of 1500# at the hot day, 2500 ft. conditions.

11. Main fan scroll area trimmed for rated EGT at hot day, 2500 ft. condition.

12. 100% RPM on the gas generator and fans is defined as follows:

100% J-85 RPM = 16,500

100% X-353-5B RPM = 2,640

100% X-376 RPM = 4,074

13. Total trimmed lift is defined as the sum of the maximum main fan lift available at 100% gas generator, with no allowance for control, and the amount of pitch fan lift required for longitudinal trim for the particular center of gravity position.

This performance is furnished as being the best estimate of the X-353-5B (main) and X-376 (pitch) fan performance at this time. It is anticipated that the X-353-5B and X-376 performance specifications will be up-dated based upon these and subsequent test data, at a future date.

E G Smith

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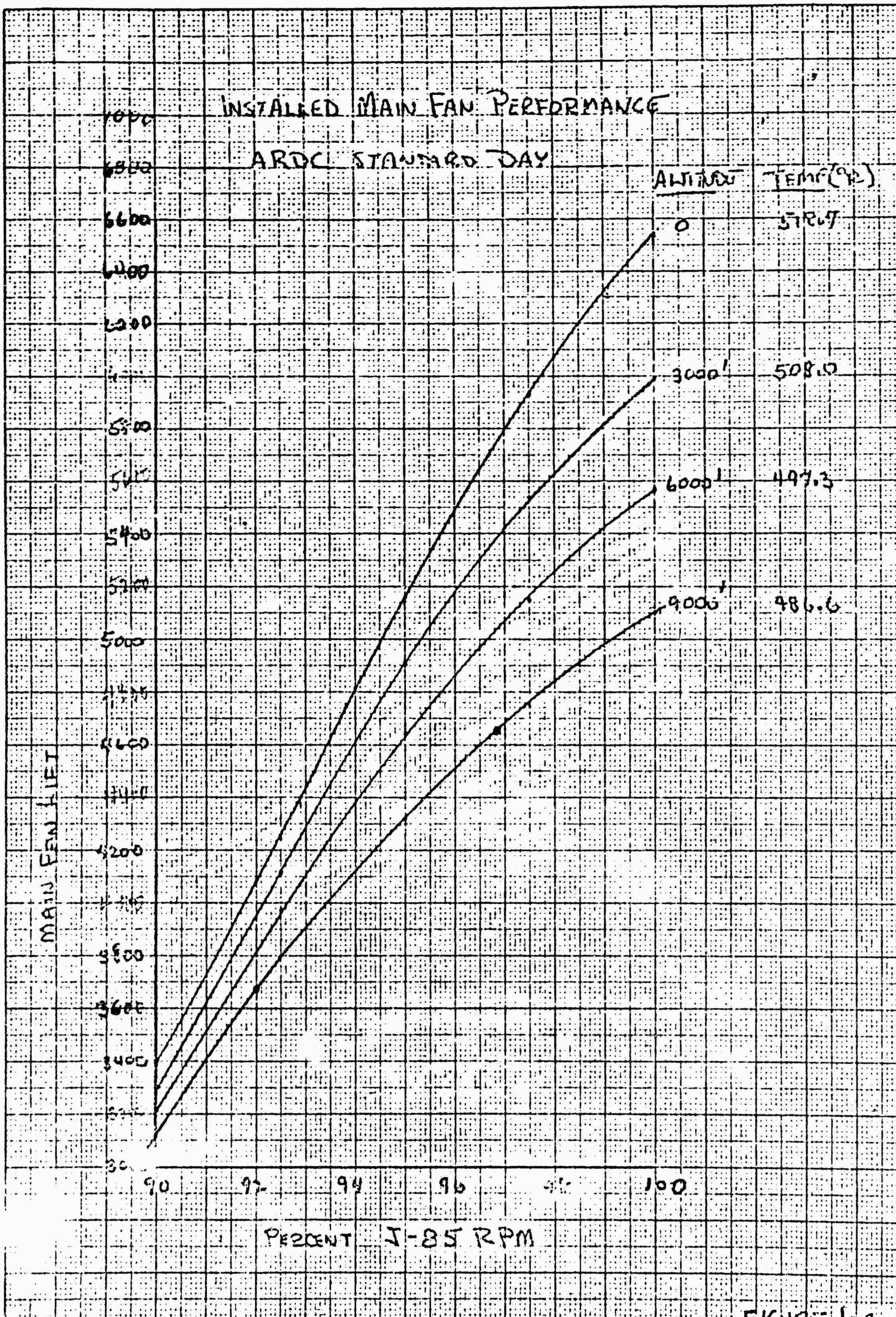
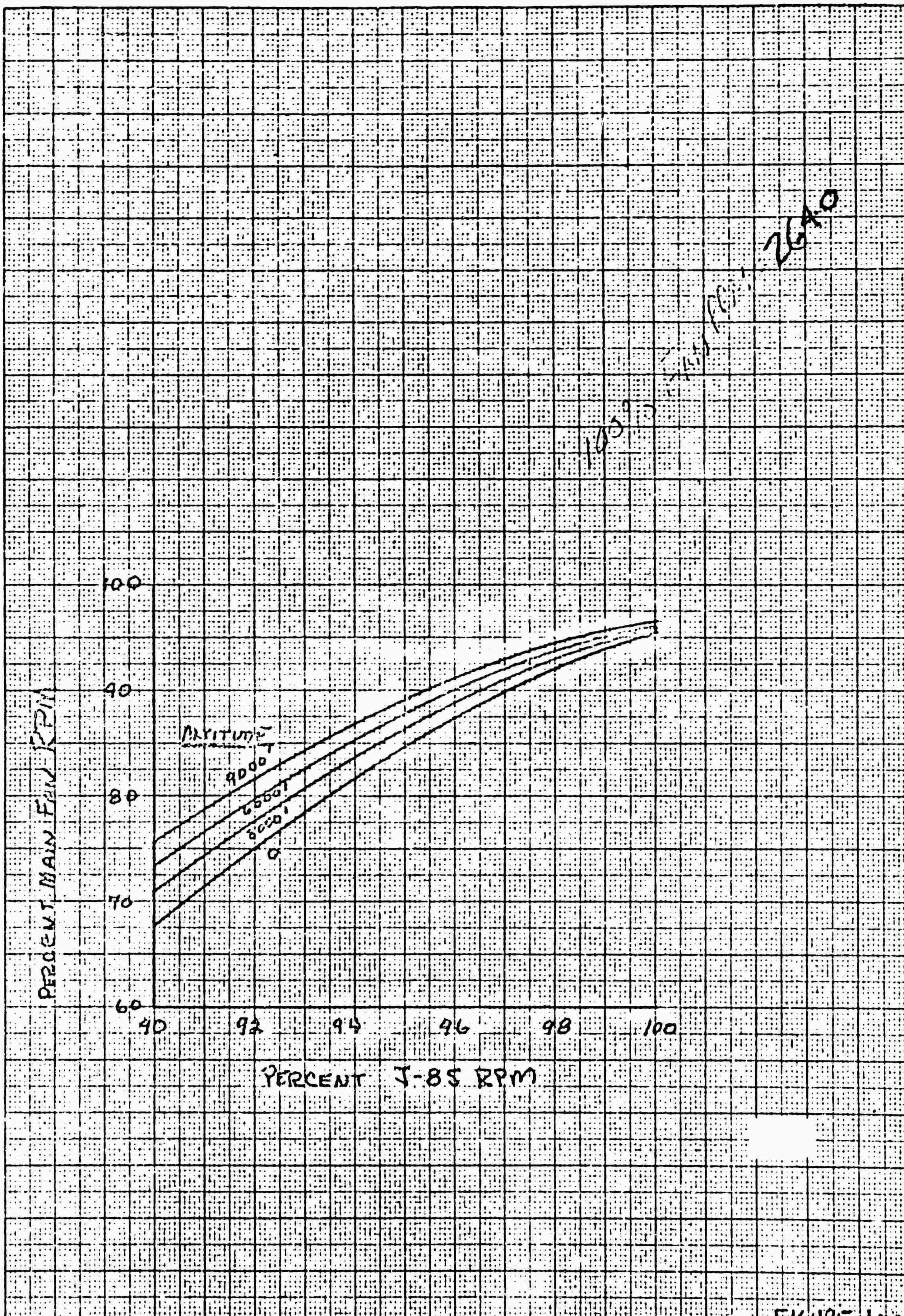
EGS/ht



INSTALLED MAIN FAN PERFORMANCE
ARDC STANDARD DAY
ALTITUDE
TEMP (°R)
0
518.7
3000'
508.0
6000'
497.3
9000'
486.6
6600
6400
6200
5800
5400
5000
4600
4200
3800
3600
3400
90
94
96
100
MAIN FAN LIFT
PERCENT J-85 RPM

FIGURE 1-a


PERCENT MAIN FAN RPM
PERCENT J-85 RPM
ALTITUDE
9000
6500
3000
0
100
90
80
70
60
90
92
94
96
98
100
FIGURE 1-5

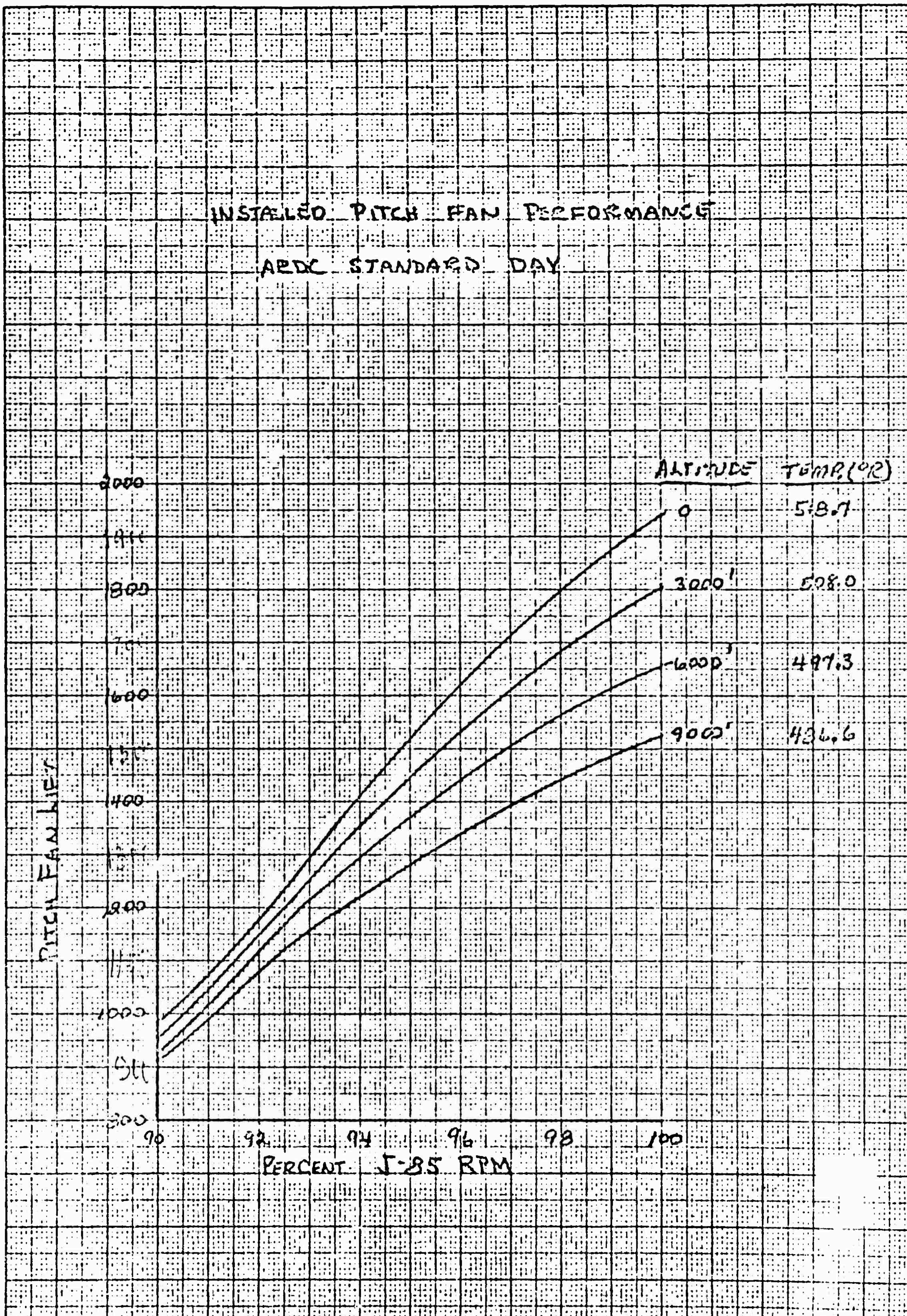


FIGURE 2-

RYAN 64B015

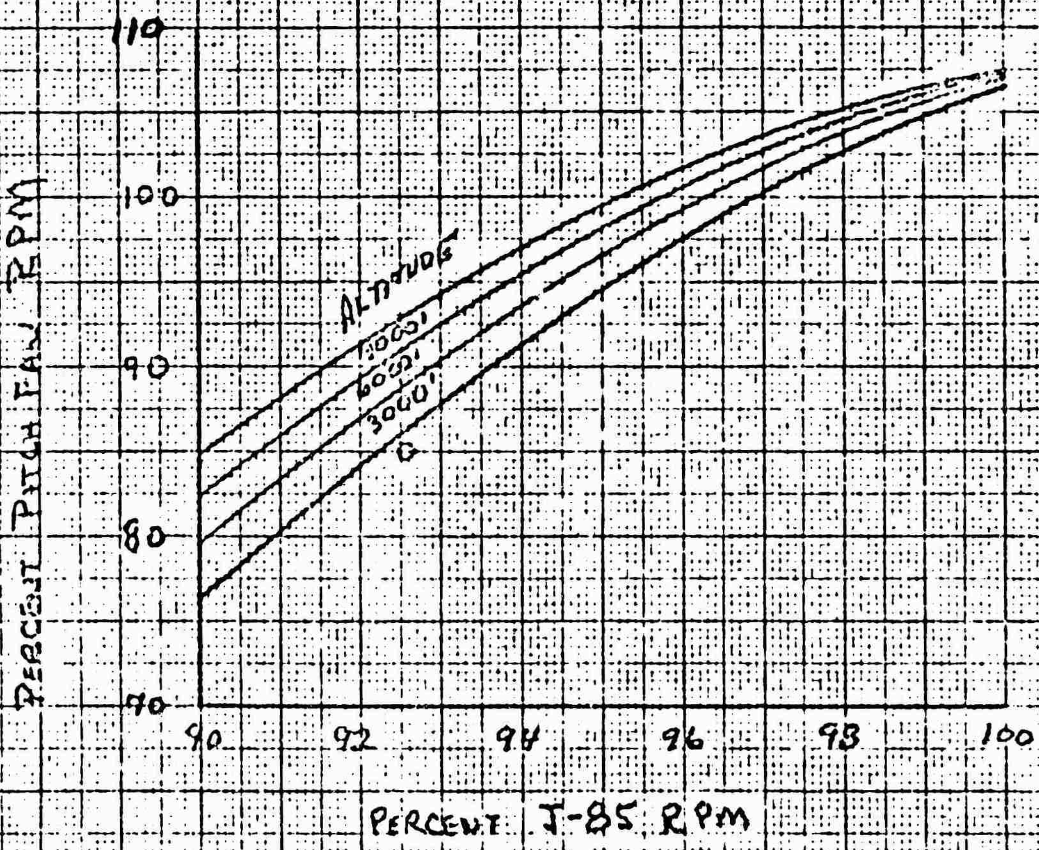

PERCENT PITCH FAN RPM
110
100
90
80
70
ALTITUDE
10000'
6000'
3000'
0
90
92
94
96
98
100
PERCENT J-85 RPM

FIGURE 2-5

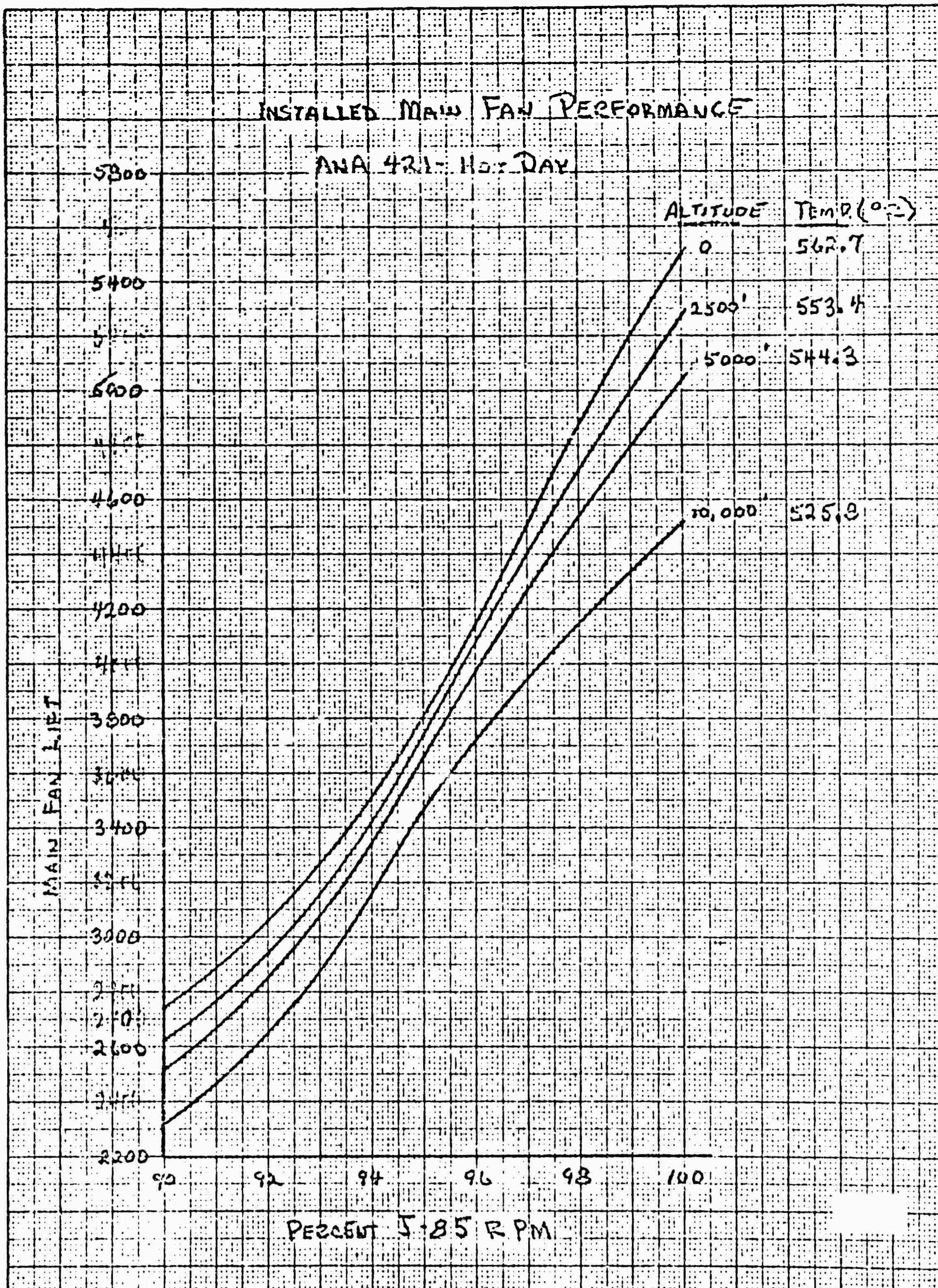
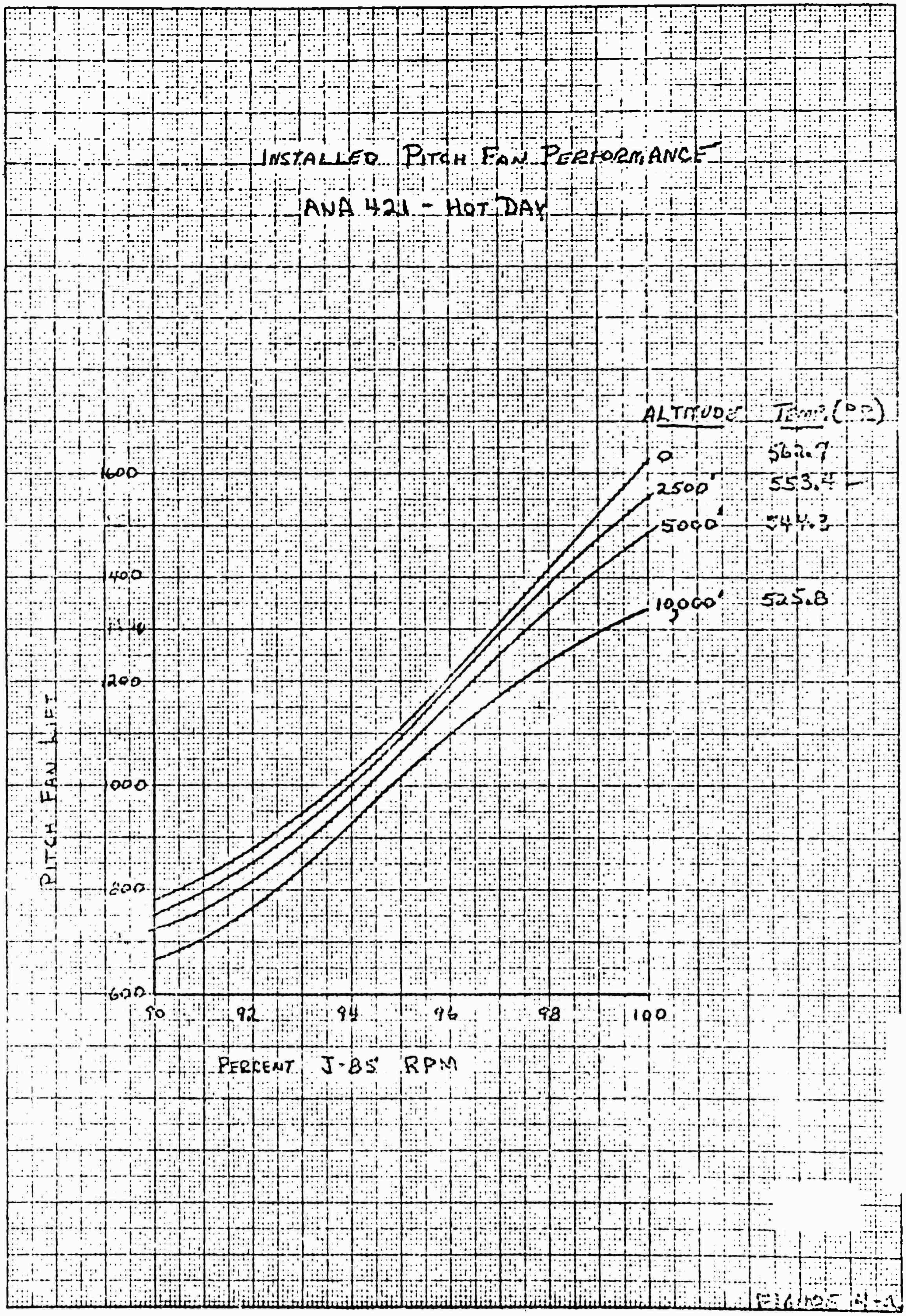

INSTALLED MAIN FAN PERFORMANCE
ANA 421 - HOT DAY
ALTITUDE
TEMP (°R)
0
562.7
2500'
553.4
5000'
544.3
10,000
525.9
5800
5400
5000
4600
4200
3800
3400
3000
2600
2200
MAIN FAN LIFT
90
92
94
96
98
100
PERCENT J-85 RPM

FIGURE 3-C



INSTALLED PITCH FAN PERFORMANCE
ANA 421 - HOT DAY
ALTITUDE
0
2500'
5000'
10,000'
562.7
553.4
525.8
PITCH FAN LIFT
1600
400
200
000
800
600
90
92
94
96
98
100
PERCENT J-85 RPM

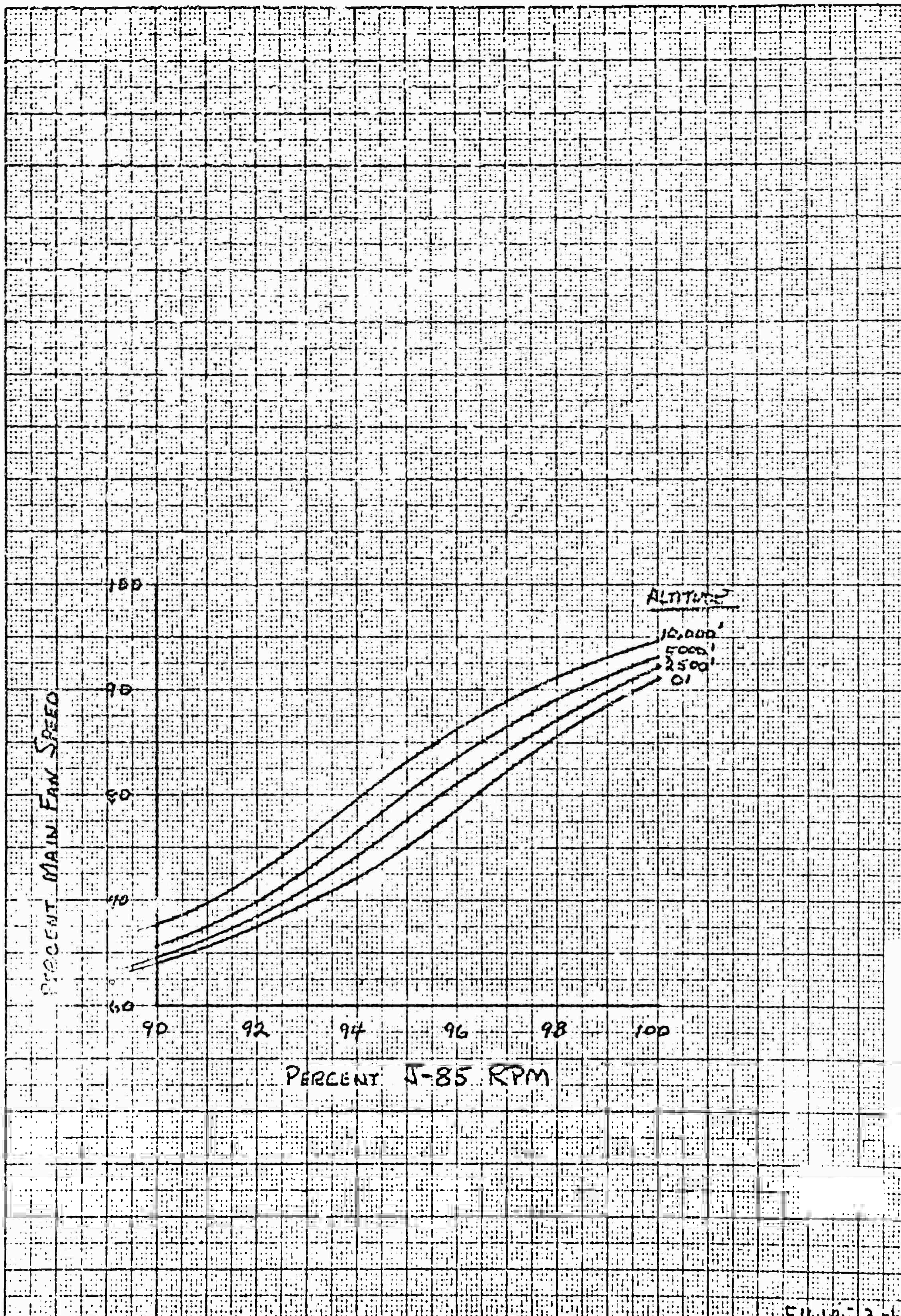
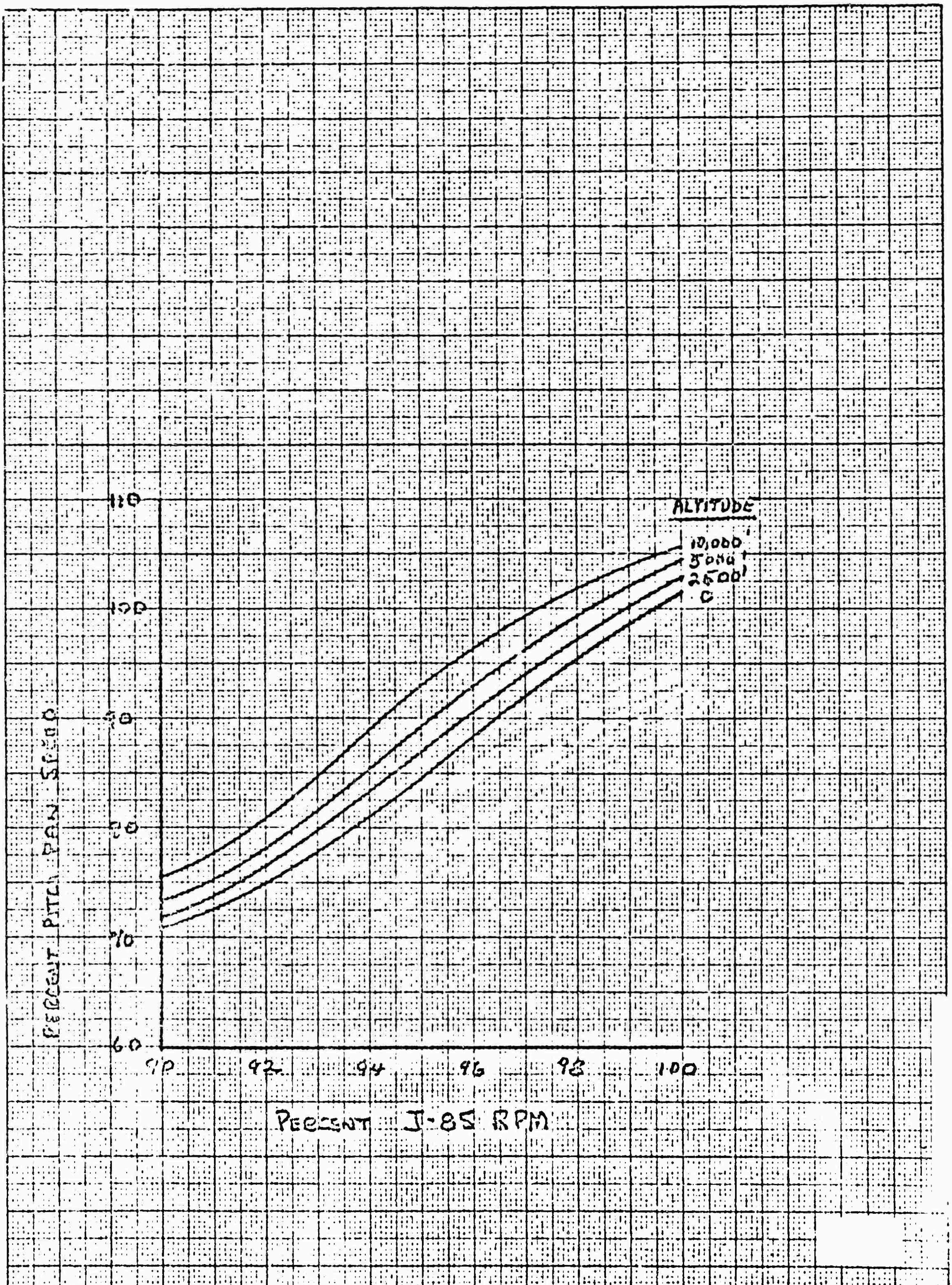


FIGURE 3-6

RYN [redacted] 64B015



ALTITUDE
10,000'
5000'
2500'
0
110
100
90
80
70
60
90
92
94
96
98
100
PERCENT PITCH FAN SPEED
PERCENT J-85 RPM

FIGURE II-b

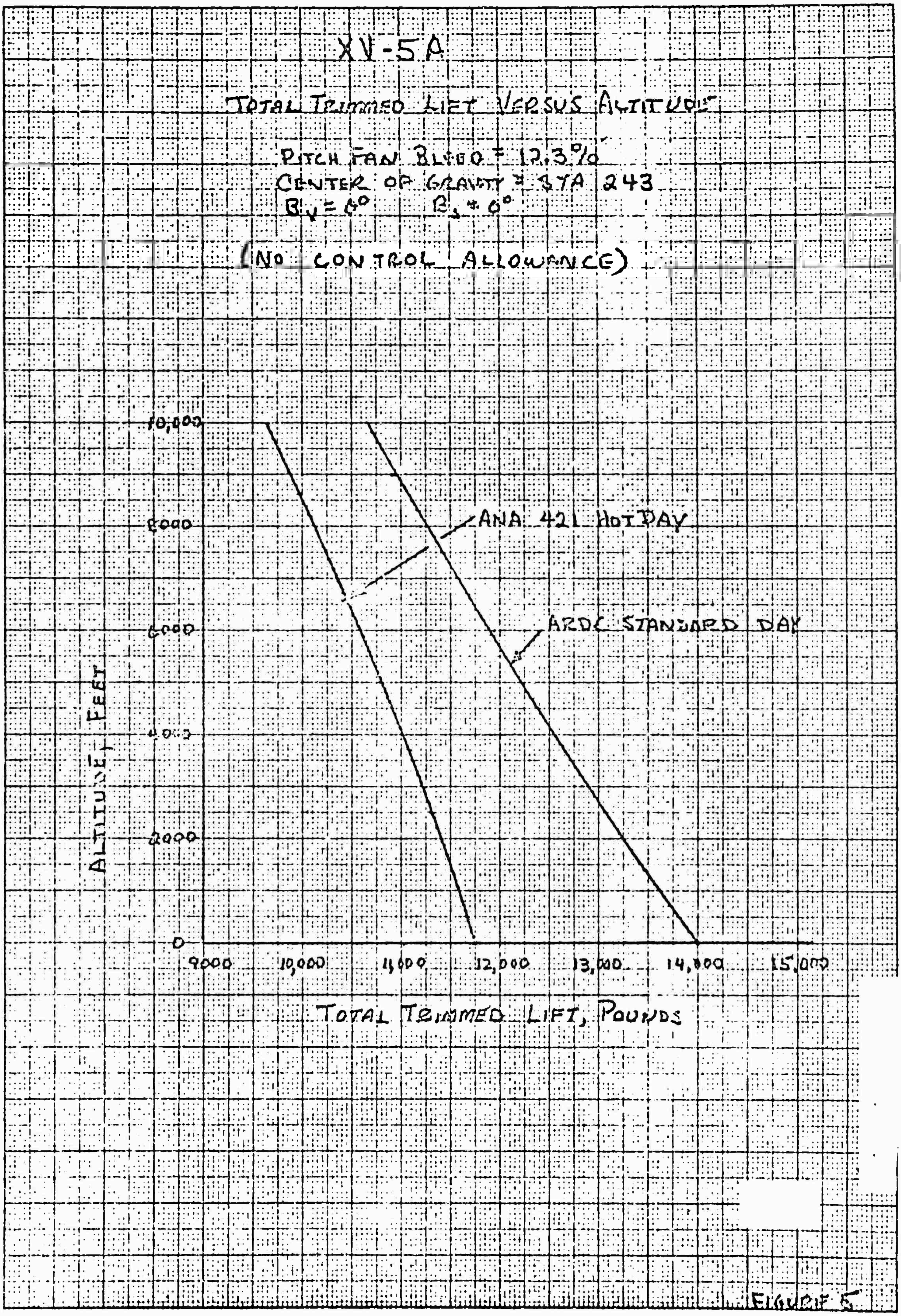


FIGURE 5

RYAN 64B015

7.3 METHOD OF ESTIMATING TURBOJET MODE INSTALLED PERFORMANCE

The method of estimating GE X353-5B turbojet mode installed performance calls upon the data of the following Tables 7.3.1 through 7.3.6 while operating upon the initial computer output of Step 3, (Section 4.1) to obtain the final installed performance in Step 4.

Tables 7.3.7 through 7.3.11 present a typical set of Step 3 output data from the computer deck (see Reference 10 of Section 7.0 of this report). Table 7.3.12 contains typical Step 4 final installed performance resulting from the special IBM routine using data of Tables 7.3.7 through 7.3.11 as input. Symbols for the above Tables are defined in Tables 7.3.13 and 7.3.14. The calculation procedure from Tables 7.3.7 through 7.3.11 and Table 7.3.12 is as follows:

1. Data of Tables 7.3.7 through 7.3.11 are key-punched to designated fields.

2. From the input data, the type of day and altitude is determined for the particular run number in question.

3. The value of M/M* is obtained from the equation $M/M^* = (1.8805)\ Wa \sqrt{T_{amb}} / A_t P_{amb} (1 + .2MP^2)^3$ where $A_t = 182.3\ in^2$ and the appropriate values of $\sqrt{T_{amb}} / P_{amb}$ and WA are read from Tables 7.3.1 through 7.3.11 respectively.

4. For the given Mach No. and number of engines operating, values of inlet recovery factor (NRI) and external drag coefficient (CDS) are read from Tables 7.3.2 and 7.3.3.

5. Steps 2 through 4 are repeated for all runs for a given aircraft and engine operating condition, to form a set of computed and assumed inlet recovery factors, (NRI) and (NRA) respectively.

6. The correct inlet recovery is then computed directly from the data of Step 5.

7. All output performance variables are then interpolated or extrapolated to obtain their proper value at the calculated inlet recovery factor of Step 6.

8. The external incremental drag (SID) is calculated by the equation SID = (127.61) (CDS) (PO) $(MO)^2$ based on the minimum area of the 30E inlet. The proper value of PO is selected from Table 7.3.1.

9. For the case of single engine operation, the windmilling drag is selected from Table 7.3.4.

10. Several arithmetical steps complete the intermediate performance before applying ejector effects:

    a. Gross thrust (FG) is obtained by adding ram drag (FD) to net thrust (FN).

    b. Propulsion system drag (DRAG) is obtained by adding ram drag (FD), incremental drag (SID), and windmilling drag (FWIND).

11. To obtain ejector effects on gross thrust, the proper value of thrust ratio (FEJ/FG) was interpolated from the data of Table 7.3.5, at the exhaust gas-to-ambient pressure ratio (P8/P0) obtained after interpolation at the proper inlet recovery factor for a pre-selected series of shroud air to ambient pressure ratios (PS/PO) thereby forming a set of data in terms of (PS/PO).

12. Selecting the appropriate value of (PS/PO) from the data of Table 7.3.6 then permitted interpolation of the gross thrust ratio and determination of gross thrust (FEJ).

13. Net thrust (FNET) was obtained by subtracting the propulsion system drag (DRAG) from the gross thrust (FEJ).

14. Since estimated performance is based on single engine performance, total aircraft performance with two engines operating is obtained by simple multiplication of the appropriate quantity by ENG=2; e.g. TDRAG = (DRAG) (ENG) = (DRAG) (2). Of course, for one engine operating, (ENG) = 1 and TDRAG = DRAG.

TABLE 7.3.1

Effect of Altitude and Day on $\sqrt{T_{amb}}/P_{amb}$ Ratio and Ambient Pressure (PO)

DAY	ALTITUDE (Feet)	$\sqrt{T_{amb}}/P_{amb}$	PO (psi)
Std.	0	1.549747	14.6958
	5,000	1.830143	12.2285
	10,000	2.174270	10.1083
	20,000	3.129630	6.7588
	30,000	4.641250	4.3726
	40,000	7.233760	2.7300
Hot	0	1.614141	14.6958
	2,500	1.741273	13.5098
	5,000	1.881008	12.4018
	10,000	2.203316	10.40[illegible]1
	20,000	3.060443	7.1862
	30,000	4.410434	4.8188
	40,000	6.520842	3.1229

TABLE 7.3.2

Estimated XV-5A 30E Inlet Performance for 2 Engine Operation and External Incremental Drag Coefficient

MO	M/M*	NRI	CDS	MO	M/M*	NRI	CDS
0	.35	.9927	----	.2	.35	.9975	----
	.4	.9925	0.00		.4	.9976	0.00
	.45	.9919	----		.45	.9975	----
	.5	.9908	0.00		.5	.9974	0.00
	.55	.9889	----		.55	.9971	----
	.6	.9867	0.00		.6	.9964	0.00
	.65	.9843	----		.65	.9953	----
	.7	.9814	0.00		.7	.9940	0.00
	.75	.9782	----		.75	.9923	----
	.8	.9747	0.00		.8	.9903	0.00
	.85	.9707	----		.85	.9880	----
	.90	.9644	0.00		.90	.9853	0.00
.1	.35	.9935	----	.3	.35	.9992	-----
	.4	.9936	0.00		.4	.9993	0.015
	.45	.9930	----		.45	.9993	-----
	.5	.9920	0.00		.5	.9992	0.000
	.55	.9904	----		.55	.9987	-----
	.6	.9887	0.00		.6	.9980	0.000
	.65	.9869	----		.65	.9971	-----
	.7	.9823	0.00		.7	.9956	0.000
	.75	.9811	----		.75	.9946	-----
	.8	.9770	0.00		.8	.9930	0.000
	.85	.9721	----		.85	.9913	-----
	.90	.9701	0.00		.95	.9889	0.000

TABLE 7.3.2 (Continued)

MO	M/M*	NRI	CDS	MO	M/M*	NRI	CDS
.4	.35	.9998	-----	0.6	.35	1.0000	-----
	.4	.9999	0.065		.4	1.0000	0.150
	.45	.9999	-----		.45	1.0000	-----
	.5	.9998	0.033		.5	1.0000	0.112
	.55	.9993	-----		.55	1.0000	-----
	.6	.9990	0.007		.6	1.0000	0.076
	.65	.9980	-----		.65	.9997	-----
	.7	.9969	0.000		.7	.9990	0.041
	.75	.9955	-----		.75	.9970	-----
	.8	.9940	0.000		.8	.9966	0.012
	.85	.9921	-----		.85	.9951	-----
	.90	.9900	0.000		.90	.9933	0.000
0.5	.35	.9999	-----	0.7	.35	.9822	-----
	.4	1.0000	0.110		.4	.9940	0.184
	.45	1.0000	-----		.45	.9972	-----
	.5	.9999	0.075		.5	.9993	0.145
	.55	.9998	-----		.55	1.0000	-----
	.6	.9994	0.044		.6	1.0000	0.104
	.65	.9990	-----		.65	.9999	-----
	.7	.9982	0.011		.7	.9997	0.067
	.75	.9967	-----		.75	.9994	-----
	.8	.9952	0.000		.8	.9988	0.034
	.85	.9934	-----		.85	.9980	-----
	.90	.9912	0.000		.90	.9968	0.001

TABLE 7.3.2 (Continued)

MO	M/M*	NRI	CDS	MO	M/M*	NRI	CDS
0.8	.35	.9220	-----				
	.4	.9503	0.216				
	.45	.9650	-----				
	.5	.9739	0.171				
	.55	.9795	-----				
	.60	.9833	0.127				
	.65	.9844	-----				
	.70	.9847	0.089				
	.75	.9846	-----				
	.80	.9842	0.051				
	.85	.9835	-----				
	.90	.9828	0.018				

TABLE 7.3.3

Estimated XV-5A 30E Inlet Performance
for 1 Engine Operation

MO	M/M*	NRI	MO	M/M*	NRI
0	.4	.9655	.2	.4	.9883
	.45	.9645		.45	.9879
	.5	.9633		.5	.9875
	.55	.9624		.55	.9865
	.6	.9609		.6	.9853
	.65	.9595		.65	.9837
	.7	.9564		.7	.9807
	.75	.9526		.75	.9752
	.8	.9489		.8	.9645
	.85	.9391		.85	.9546
	.90	.9262		.90	.9409
.1	.4	.9732	.3	.4	.9964
	.45	.9713		.45	.9963
	.5	.9698		.5	.9962
	.55	.9685		.55	.9959
	.6	.9670		.6	.9955
	.65	.9653		.65	.9947
	.7	.9625		.7	.9939
	.75	.9580		.75	.9925
	.8	.9535		.8	.9904
	.85	.9447		.85	.9870
	.90	.9315		.95	.9660

TABLE 7.3.3 (Continued)

MO	M/M*	NRI	MO	M/M*	NRI
.4	.4	.9986	.6	.4	.9764
	.45	.9986		.45	.9814
	.5	.9985		.5	.9858
	.55	.9983		.55	.9894
	.6	.9980		.6	.9922
	.65	.9976		.65	.9942
	.7	.9971		.7	.9952
	.75	.9965		.75	.9953
	.8	.9954		.8	.9951
	.85	.9944		.85	.9944
	.90	.9926		.90	.9928
.5	.4	.9940	.7	.4	.9310
	.45	.9959		.45	.9450
	.5	.9964		.5	.9578
	.55	.9966		.55	.9689
	.6	.9973		.6	.9784
	.65	.9975		.65	.9863
	.7	.9970		.7	.9921
	.75	.9961		.75	.9948
	.8	.9956		.8	.9943
	.85	.9950		.85	.9916
	.90	.9943		.90	.9873

TABLE 7.3.4

Estimated Windmilling Drag for 1 Engine Out Condition
(Pounds)

ARDC STD. DAY

MP/ALT	0	2,500	5,000	10,000	20,000	30,000	40,000
0	0	-----	0	0	0	0	0
.1	3.7	-----	3.1	2.5	1.7	1.1	.7
.2	11.5	-----	9.6	7.9	5.3	3.4	2.1
.3	26.5	-----	22.1	18.2	12.2	7.9	4.9
.4	50.0	-----	41.6	34.4	23.0	14.9	9.3
.5	83.5	-----	69.5	57.4	38.4	24.8	15.5
.6	127.5	-----	106.1	87.7	58.6	37.9	23.7
.7	182.0	-----	151.4	125.2	83.7	54.2	33.8
.8	245.5	-----	204.3	168.9	112.7	73.0	45.8
ANA BUL. 421 HOT DAY							
0	0	0	0	0	0	0	0
.1	3.4	3.1	2.9	2.4	1.7	1.1	0.7
.2	10.6	9.7	8.9	7.5	5.2	3.5	2.3
.3	24.4	22.4	20.6	17.3	11.9	8.0	5.2
.4	46.1	42.4	38.9	32.6	22.5	15.1	9.8
.5	77.0	70.8	65.0	54.5	37.7	25.2	16.4
.6	117.5	108.0	99.2	83.2	57.5	38.5	25.0
.7	167.8	154.3	141.6	118.8	82.1	55.0	35.7
.8	226.3	208.0	191.0	160.2	110.7	74.2	48.1

TABLE 7.3.5

Estimated Ejector Performance Affecting Gross Thrust

PS/PO	P8/PO	FEJ/FJ	PS/PO	P8/PO	FEJ/FJ
0.8	1.00	.968	0.9	3.10	.981
	1.50	.968		3.20	.979
	2.00	.968		3.30	.975
	2.28	.968		3.40	.971
	2.30	.969		3.43	.968
	2.40	.971		10.0	.968
	2.50	.973	.95	1.0	.9895
	2.6	.976		2.08	.9895
	2.7	.978		2.20	.9895
	2.8	.979		2.40	.9905
	2.9	.975		2.6	.9925
	3.0	.962		2.8	.9925
	3.02	.954		3.0	.9915
	10.0	.954		3.2	.989
0.9	1.00	.982		3.4	.984
	2.18	.982		3.6	.975
	2.20	.983		3.65	.9725
	2.30	.9835		10.	.9725
	2.40	.984	1.0	1.0	.990
	2.50	.9850		2.05	.990
	2.60	.9855		2.20	.9915
	2.70	.986		2.50	.998
	2.80	.985		3.00	1.001
	2.90	.984		3.50	.9925
	3.00	.983		3.66	.989

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TABLE 7.3.5 (Continued)

PS/PO	P8/PO	FEJ/FJ	PS/PO	P8/PO	FEJ/FJ
1.0	3.85	.978	1.20	1.000	1.059
	10.00	.978		1.011	1.059
				1.40	1.055
1.05	1.00	.997		1.90	1.049
	2.01	.997		2.4	1.040
	2.1	1.000		2.9	1.031
	2.5	1.007		3.4	1.021
	3.0	1.009		3.9	1.009
	3.4	1.001		4.4	.999
	3.9	.991		4.6	.991
	4.06	.982		10.0	.991
	10.0	.982			
1.10	1.00	1.014			
	1.70	1.014			
	1.90	1.017			
	2.40	1.0195			
	2.50	1.020			
	2.90	1.018			
	3.40	1.011			
	3.90	1.002			
	4.00	.997			
	4.22	.985			
	10.	.985			

TABLE 7.3.6

Estimated Secondary Ejector Air to Ambient Pressure Ratio

A/C MACH NO.	RPM 100%	RPM 99%	RPM 98%	RPM 97%	RPM 95%	RPM 90%	RPM 85%	RPM 80%	RPM 75%
0	.957	.964	.967	.968	.970	.965	.961	.963	.969
.10	.964	.971	.973	.974	.975	.969	.965	.968	.978
.20	.976	.979	.980	.981	.982	.981	.979	.980	.988
.30	.993	.993	.994	.994	.994	.993	.993	.995	1.001
.40	1.014	1.010	1.009	1.008	1.007	1.008	1.010	1.012	1.016
.50	1.038	1.030	1.027	1.025	1.023	1.021	1.022	1.025	1.031
.60	1.065	1.050	1.044	1.039	1.035	1.032	1.034	1.037	1.047
.70	1.097	1.066	1.058	1.053	1.047	1.040	1.042	1.048	1.064
.80	1.138	1.089	1.078	1.070	1.060	1.050	1.049	1.055	1.083

TABLE 7.3.7

Representative Initial Installed Performance Output from GE X353-5B Turbojet Mode Computer Deck vs NRA

XV-5A FINAL INSTALLED PERFORMANCE
ARDC STANDARD DAY, DP6=0.031, A8=111.54
ALTITUDE 00000 NRA=0.980 MP=0.0

GENERAL ELECTRIC X-353-5B TURBOJET MODE PERFORMANCE

MILITARY RATED PERFORMANCE

INPUT

I	P.S.	PO	TO
1	4.000	14.696	518.69
WEB	HP	P2	T2
0.022	40.000	14.402	518.69
DELTO	ALT.	MP	RPMI
0.0			100.0

OUTPUT

ERI	FN	FD	SFC
0	2479.08	0.00	1.042
HRGT	PCT. N	PEB	TEB
1689.0	100.00	87.80	946.0
WIB	FNIB	PIB	TIB
0.0000	0.00	0.000	0.0
WA	AE8	P8/PO	T8
42.84	109.6	2.02	1675.2
WF	P6	W6	
2583.7	30.64	42.76	

DATED JANUARY 1962

TABLE 7.3.8

Representative Initial Installed Performance Output from GE X353-5B Turbojet Mode Computer Deck vs NRA

XV-5A FINAL INSTALLED PERFORMANCE
ARDC STANDARD DAY, DP6=0.031, A8=111.54
ALTITUDE 00000 NRA=0.985 MP=0

GENERAL ELECTRIC X353-5B TURBOJET MODE PERFORMANCE

MILITARY RATED PERFORMANCE

INPUT

I	P.S.	PO	TO
2	4.000	14.696	518.69
WEB	HP	P2	T2
0.022	40.000	14.475	518.69
DELTO	ALT.	MP	RPMI
0.0			100.0

OUTPUT

ERI	FN	FD	SFC
0	2499.27	0.00	1.039
HRGT	PCT. N	PEB	TEB
1688.7	100.00	88.24	945.9
WIB	FNIB	PIB	TIB
0.0000	0.00	0.000	0.0
WA	AE8	P8/PO	T8
43.06	109.6	2.03	1674.9
WF	P6	W6	
2596.1	30.79	42.98	

DATED JANUARY 1962

TABLE 7.3.9

Representative Initial Installed Performance Output from GE X353-5B Turbojet Mode Computer Deck

XV-5A FINAL INSTALLED PERFORMANCE
ARDC STANDARD DAY, DP6-0.031, A8-111.54
ALTITUDE 00000 NRA=0.990 MP=0

GENERAL ELECTRIC X353-5B TURBOJET MODE PERFORMANCE

MILITARY RATED PERFORMANCE

INPUT

I	P.S.	PO	TO
3	4.000	14.696	518.69
WEB	HP	P2	T2
0.022	40.000	14.549	518.69
DELTO	ALT.	MP	RPMI
0.0			100.0

OUTPUT

ERI	FN	FD	SFC
0	2519.38	0.00	1.035
HRGT	PCT. N	PEB	TEB
1688.3	100.00	88.68	945.9
WIB	FNIB	PIB	TIB
0.0000	0.00	0.000	0.0
WA	AE8	P8/PO	T8
43.28	109.6	2.04	1674.6
WF	P6	W6	
2608.5	30.94	43.20	

DATED JANUARY 1962

TABLE 7.3.10

Representative Initial Installed Performance Output from GE X353-5B Turbojet Mode Computer Deck

XV-5A FINAL INSTALLED PERFORMANCE
ARDC STANDARD DAY, DP6=0.031, A8=111.54
ALTITUDE 00000 NRA=0.995 MP=0

GENERAL ELECTRIC X353-5B TURBOJET MODE PERFORMANCE

MILITARY RATED PERFORMANCE

INPUT

I	P.S.	PO	TO
4	4.000	14.696	518.69
WEB	HP	P2	T2
0.022	40.000	14.622	518.69
DELTO	ALT.	MP	RPMI
0.0			100.0

OUTPUT

ERI	FN	FD	SFC
0	2539.78	0.00	1.032
HRGT	PCT. N	PEB	TEB
1688.1	100.00	89.13	945.9
WIB	FNIB	PIB	TIB
0.0000	0.00	0.000	0.0
WA	AE8	P8/PO	T8
43.50	109.6	2.05	1674.4
WF	P6	W6	
2621.2	31.10	43.42	

DATED JANUARY 1962

TABLE 7.3.11

Representative Initial Installed Performance Output from GE X353-5B Turbojet Mode Computer Deck

XV-5A FINAL INSTALLED PERFORMANCE
ARDC STANDARD DAY, DP6=0.031, A8=111.54
ALTITUDE 00000 NRA=1.000 MP=0

GENERAL ELECTRIC X353-5B TURBOJET MODE PERFORMANCE

MILITARY RATED PERFORMANCE

<u>INPUT</u>

I	P.S.	PO	TO
5	4.000	14.696	518.69
WEB	HP	P2	T2
0.022	40.000	14.696	518.69
DELTO	ALT.	MP	RPMI
0.0			100.0

<u>OUTPUT</u>

ERI	FN	FD	SFC
0	2559.88	0.00	1.029
HRGT	PCT. N	PEB	TEB
1687.8	100.00	89.57	945.9
WIB	FNIB	PIB	TIB
0.0000	0.00	0.000	0.0
WA	AE8	P8/PO	T8
43.72	109.6	2.06	1674.0
WF	P6	W6	
2633.5	31.26	43.64	

DATED JANUARY 1962

TABLE 7.3.12

Representative XV-5A Installed Performance Output for GE X353-5B Turbojet Mode

XV-5A FINAL INSTALLED PERFORMANCE (TURBOJET MODE) DATE 4/27/64

INPUT

A8	DP6	WEB	HP	DAY	ENG	PCT.N	ALT	MO
111.54	0.031	0.022	40.0	2	2	100.0	0.	0.

RUN	ERI	NRA	HRGT	FN	FD	FNIB	WF	WA	P8/P0
1	0	0.9800	1689.0	2479.08	0.	0.	2583.7	42.84	2.02
2	0	0.9850	1688.7	2499.27	0.	0.	2596.1	43.06	2.03
3	0	0.9900	1688.3	2519.38	0.	0.	2608.5	43.28	2.04
4	0	0.9950	1688.1	2539.78	0.	0.	2621.2	43.50	2.05
5	0	1.0000	1687.8	2559.88	0.	0.	2633.5	43.72	2.06

OUTPUT

NRI	HRGT	FN	FD	FNIB	WF	WA	P8/P0
0.9847	1688.9	2488.13	-0.	-0.	2589.2	42.94	2.02

M/M*	CDS	TFD	PO	TWF	SID	FDWIND
0.6861	0.	-0.	14.6958	5178.5	0.	0.

TABLE 7.3.12 (Continued)

PS/PO	FEJ/FG	FEJ	TFEJ	FNET	TFNET
0.8000	0.9680	2408.51	4817.01	2408.51	4817.01
0.9000	0.9820	2443.34	4886.68	2443.34	4886.68
0.9500	0.9890	2460.76	4921.51	2460.76	4921.51
1.0000	0.9900	2463.24	4926.49	2463.24	4926.49
1.0500	0.9975	2481.95	4963.90	2481.95	4963.90
1.1000	1.0176	2531.97	5063.95	2531.97	5063.95
1.2000	1.0468	2604.47	5208.94	2604.47	5208.94

PS/PO	FEJ/FG	FEJ	TFEJ	FNET	TFNET	DRAG	TDRAG
0.9570	0.9887	2460.13	4920.26	2460.13	4920.26	-0.	-0.

NOTE -- NRI IS WITHIN NRA BOUNDS

TABLE 7.3.13

Definition of Symbols for Tables 7.3.7 through 7.3.11 (in Order of their Appearance)

SYMBOL	MEANING
DP6	% Pressure Loss in Exhaust Duct/100
A8	Exhaust Nozzle Area in^2
NR	Assumed Inlet Recovery Factor
MP	Mach No.
I	Run Number
P.S.	Power Setting
PO	Ambient Pressure PSIA
TO	Ambient Temperature °R
WEB	Customer Compressor Bleed Air lbs/sec.
HP	Power Extraction HP
P2	Total Pressure at Compressor Face PSIA
T2	Total Temperature at Compressor Face
DELTO	(Not used as input)
Alt	(Not used as input)
RPMI	% RPM of Gas Generator
ERI	Error Indicator
FN	Net Thrust lbs.
FD	Ram Drag lbs.
SFC	Specific Fuel Consumption (WF)/(FN) = (lbs./hr)/lbs Thrust
HRGT	Harness Gas Temperature °R
PCT N	% RPM Gas Generator
PEB	Customer Compressor Bleed Air Pressure PSIA
TEB	Customer Compressor Bleed Air Temperature °R
WIB	Interstage Bleed Air lbs/sec

TABLE 7.3.13 (Continued)

SYMBOL	MEANING
FNIB	Ideal Thrust Obtainable from Interstage Bleed Air
PIB	Interstage Bleed Air Pressure PSIA
TIB	Interstage Bleed Air Temperature °R
WA	Engine Inlet Air lbs/sec
AE8	Effective Tailpipe Area in^2
P8/PO	Tailpipe to Ambient Pressure Ratio
T8	Actual Tailpipe Temperature
WF	Fuel Flow lbs/hr
P6	Pressure at Outlet of Diverter Valve PSIA
W6	Exhaust Gas after Diverter Valve lbs/sec

TABLE 7.3.14

Definitions of Symbols for Table 7.3.12 (in Order of their Appearance)

SYMBOL	MEANING
A8	Tailpipe Nozzle Area-in^2
DP6	% Pressure Loss in Exhaust Duct
WEB	Customer Compressor Bleed Air - lbs/sec
HP	Power Extraction - HP
DAY	ARDC Standard Day (2), ANA Bulletin 421 Hot Day (1)
ENG	Number of Engines Operating
PCT.N	% RPM Gas Generator
Alt.	Altitude - feet
MO	Mach No.
RUN	Run No. of Table 7-7
ERI	Error Indicator
NRA	Assumed Inlet Pecovery Factor

TABLE 7.3.14 (Continued)

SYMBOL	MEANING
HRGT	Harness Exhaust Gas Temperature -- °R
FN	Net Thrust - lbs.
FD	Ram Drag - lbs.
FNIB	Ideal Thrust Possible from Interstage Bleed
WF	Fuel Flow - lbs/hr.
WA	Engine Inlet Air - lbs/sec.
P8/PO	Tailpipe to Ambient Pressure Ratio
NRI	Computed Inlet Recovery Factor
M/M*	Inlet Mass Flow Ratio
CDS	External Incremental Drag Coefficient
TFD	Total Ram Drag for Engines Operating - lbs.
PO	Ambient Pressure - PSIA
TWF	Total Fuel Flow for Engines Operating - lbs/hr.
SID	External Incremental Drag - lbs.
FDWIND	Windmilling Drag - lbs.
PS/PO	Secondary Ejector Air to Ambient Pressure Ratio
FEJ/FG	Gross Thrust ratio with and without Ejector - lbs.
FEJ	Gross Thrust per Engine with Ejector - lbs.
TFEJ	Total Gross Thrust with Ejector for Engines Operating - lbs.
FNET	Net Thrust per Engine with Ejector - lbs. (Does not account for Cooling Air Drag)
TFNET	Total Net Thrust with Ejector for Engines Operating - lbs. (Does not account for Cooling Air Drag)
DRAG	Propulsion System Drag per Engine - lbs. (Does not account for Cooling Air Drag)
TDRAG	Total Propulsion System Drag for Engines Operating (Does not include Cooling Air Drag)